REPORT NO. F694
31 AUGUST 1967

PART G SUBSYSTEM/EQUIPMENT DESCRIPTIONS

VOYAGER CAPSULE PHASE B FINAL REPORT

VOLUME IV ENTRY SCIENCE PACKAGE

PREPARED FOR:
CALIFORNIA INSTITUTE OF TECHNOLOGY
JET PROPULSION LABORATORY
PASADENA, CALIFORNIA
CONTRACT NUMBER 952000

COPY NO. 47

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REPORT ORGANIZATION

VOYAGER PHASE B FINAL REPORT

The results of the Phase B VOYAGER Flight Capsule study are organized into several volumes. These are:

Volume I	Summary
Volume II	Capsule Bus System
Volume III	Surface Laboratory System
Volume IV	Entry Science Package
Volume V	System Interfaces
Volume VI	Implementation

This volume, Volume IV, describes the McDonnell Douglas selected design for the Entry Science Package. It is arranged in 11 parts, A through K, and bound in 4 separate documents, as noted below.

Part A	Introduction and Summary	
Part B	Objectives and Requirements	
Part C	Design Criteria and Constraints	1 Document
Part D	Selected Design Concept	
Part E	Alternatives and Systems Analysis	
Part F	Future Mission Options	1 Document
Part G	Subsystem Equipment	1 Document
Part H	Reliability	
Part I	Planetary Quarantine	
Part J	Operational Support Equipment	1 Document
Part K	Interface Alternatives	

In order to assist the reader in finding specific material relating to the Entry Science Package, Figure 1 cross indexes broadly selected subject matter, at the system and subsystem level, through all volumes.

VOLUME IV CROSS REFERENCE INDEX

ITEM / VOLUME IV PARTS		PART A INTRODUCTION AND SUMMARY	PART B OBJECTIVES AND REQUIREMENTS	PART C DESIGN CRITERIA AND CONSTRAINTS	PART D DESCRIPTION OF PREFERRED CONFIGURATION	PART E PRINCIPAL ALTERNATIVES AND SELECTION FACTORS
ESP ASPECT						
MISSION	Objectives	Sec. 1.0	1.0	–	–	–
	Operations (& Profiles)	2.0 6.3	2.0	–	2.0–d 4.1 5.0	5.1 5.8
DESIGN	Configuration	3.0 6.2	√	√	1.0	1.1
	Functional	4.0 5.0 6.0	√	√	1.0 2.0 3.0 4.0	1.0 2.0 3.0 4.0 5.0
	Weight	3.0 4.0	–	√	1.4 2.0	3.0 d
Interfaces (See Also Volume V)		Introduction	–	1.0 2.0 3.0	1.3 2.0–d 3.0–d 4.0 5.3.4	1.1 3.0–d 4.1.3 5.0
Implementation (See Also Volume VI		–	–	–	6.0	–
ESP EQUIPMENT OR SUBSYSTEM						
Imaging		Sec. 2.0 4.0 6.0	1.2 2.2	6.1	1.2.1 2.1 4.3.2.4	1.1 2.2 3.2 5.3.2.6
Atmospheric Properties Measurement		2.0 4.0 6.0	1.1	6.2	1.2.2 2.0 4.0–d	1.1 2.1 3.1 5.0–d
Engineering Instrumentation		–	–	–	Figure 3.0–1	Figure 4.2–3 4.2.2.2
Data Storage		6.0	–	–	2.5.3 3.2.1	4.2.5
Telecommunications		5.1	–	6.3	2.5.3 3.2	4.2
Power		5.2	–	6.3	2.5.1 3.1	4.1
Structural/Mechanical		–	–	4.0	2.5.4 3.3	4.3
Cabling and Packaging		–	–	5.0 6.3	2.5.1 3.4	4.4
Thermal Control		5.3	–	6.3	2.5.7 3.5 4.3.2	4.5

√ Denotes that the part or section generally applies to the topic.
d Denotes that the topic is distributed throughout the part or section.

Figure 1

PART F FUTURE MISSION OPTIONS	PART G DETAILED DESCRIPTION OF ESP EQUIPMENT	PART H RELIABILITY	PART I PLANETARY QUARANTINE	PART J OPERATIONAL SUPPORT EQUIPMENT	PART K INTERFACE ALTERNATIVES
√	–	–	–	–	–
√	d	2.3.3 3.1.1	2.0	4.3 4.4 4.5 8.0	–
–	d	–	3.0	OSE Config	1.0 2.0
√	d	1.0 2.0 3.3	–	OSE Functional Design	1.0 2.0
–	d	2.3.2	–	–	–
–	d	–	–	√	1.0 2.0 3.0
–	–	–	d	–	1.0 3.0
√	1.1	3.3	3.2	5.5	1.2
√	1.2 1.3 1.4 1.5	3.3	–	5.5	1.2
–	2.0	3.3	–	d	–
–	3.0	3.3	3.4	5.4	–
√	4.0 5.0 6.0	3.3	–	5.4	2.0
–	7.0	3.3	3.4	5.3 4.4.8	2.0
–	8.0	–	3.1	–	–
–	9.0	–	–	–	–
–	10.0	3.3	3.3	5.6	2.0

TABLE OF CONTENTS

	Page
PART G SUBSYSTEM/EQUIPMENT FUNCTIONAL DESCRIPTIONS	1
SECTION 1 EXPERIMENT INSTRUMENTS	1-1
1.1 Entry Television	1-1
1.2 Accelerometer	1-1
1.3 Pressure Transducers	1-15
1.4 Temperature Transducers	1-23
1.5 Mass Spectrometer	1-29
SECTION 2 INSTRUMENTATION EQUIPMENT	2-1
2.1 Equipment Identification and Usage	2-1
2.2 Design Requirements and Constraints	2-1
2.3 Physical Characteristics	2-4
2.4 Operational Description	2-6
2.5 Performance Characteristics	2-8
2.6 Interface Definition	2-9
2.7 Reliability Considerations	2-9
2.8 Test	2-9
2.9 Development Requirements	2-9
SECTION 3 DATA STORAGE SUBSYSTEM	3-1
3.1 Entry Science Package Data Storage Assembly	3-1
3.2 Spacecraft Mounted Entry Science Package Support Data Storage	3-13
SECTION 4 TELEMETRY SUBSYSTEM	4-1
4.1 Equipment Identification and Usage	4-1
4.2 Design Requirements and Constraints	4-4
4.3 Physical Characteristics	4-4
4.4 Operational Description	4-4
4.5 Performance	4-13
4.6 Interfaces	4-15
4.7 Reliability and Safety Considerations	4-15
4.8 Test	4-15
4.9 Development Status	4-20
SECTION 5 ENTRY SCIENCE PACKAGE RADIO SUBSYSTEM	5-1
5.1 Entry Science Package Radio	5-1
5.2 Spacecraft-Mounted ESP Support Radio	5-12

SECTION 6 ENTRY SCIENCE PACKAGE ANTENNA SUBSYSTEM 6-1
6.1 Equipment Identification and Usage 6-1
6.2 Design Requirements and Constraints 6-1
6.3 Physical Characteristics 6-3
6.4 Performance Characteristics 6-3
6.5 Reliability and Safety Considerations 6-6
6.6 Test 6-6
6.7 Development Requirements 6-8
SECTION 7 ELECTRICAL POWER SUBSYSTEM 7-1
7.1 Equipment Identification and Usage 7-1
7.2 Design Requirements and Constraints 7-3
7.3 Physical Characteristics 7-3
7.4 Operational Description 7-3
7.5 Performance Characteristics 7-6
7.6 Interface Definition 7-6
7.7 Reliability Considerations 7-9
7.8 Testing 7-13
7.9 Development Requirements 7-13
SECTION 8 STRUCTURAL/MECHANICAL SUBSYSTEM 8-1
8.1 Equipment Identification and Usage 8-1
8.2 Design Requirements and Constraints 8-2
8.3 Physical Characteristics 8-2
8.4 Operational Description 8-3
8.5 Performance Objectives 8-6
8.6 Interface Definition 8-6
8.7 Reliability and Safety Considerations 8-6
8.8 Test Requirements 8-6
8.9 Development Requirements 8-6
SECTION 9 PACKAGING AND CABLING 9-1
9.1 Equipment Identification and Usage 9-1
9.2 Design Requirements and Constraints 9-1
9.3 Physical Characteristics 9-1
9.4 Operational Description 9-4
9.5 Reliability and Safety Considerations 9-4
9.6 Development Requirements 9-4

SECTION 10	THERMAL CONTROL SUBSYSTEM	10-1
10.1	Equipment Identification and Usage	10-1
10.2	Design Requirements and Constraints	10-3
10.3	Physical Characteristics	10-4
10.4	Operational Description	10-4
10.5	Performance Objectives	10-4
10.6	Interface Definition	10-5
10.7	Reliability Considerations	10-5
10.8	Test Requirements	10-6
10.9	Development Requirements	10-7

This Document Includes the Following Pages:

Title Page

i through v

Part G 1 through 1

1-1 through 1-38

2-1 through 2-9

3-1 through 3-21

4-1 through 4-20

5-1 through 5-32

6-1 through 6-8

7-1 through 7-16

8-1 through 8-6

9-1 through 9-6

10-1 through 10-7

PART G

SUBSYSTEM/EQUIPMENT FUNCTIONAL DESCRIPTIONS

This part contains functional descriptions of the subsystems that comprise the preferred design concept of the Entry Science Package and therefore supplement the information provided in Part D. Subsections have been written, in some instances, below the subsystem level, to make a more comprehensive presentation.

An attempt has been made to maintain uniformity in the format; however, organization and content deviations occur where it is considered beneficial to the presentation. For example, Section 8.0, the Structural/Mechanical Subsystem, being essentially passive, does not lend itself to being described in functional terms; hence, more emphasis was placed on physical description.

Certain information has been provided which would ordinarily be considered outside the scope of a purely "Functional" Description. Where our analysis and investigation has led to a design incorporating well characterized or existing equipment, the details of such equipment have been included to help substantiate its selection. Also, reliability analysis has been included for convenience in relating it to the equipment design (via functional block diagrams) for which it was derived. A brief summary of the development status of the design is presented at the end of each section for convenience in relating it to the preferred design.

SECTION 1

EXPERIMENT INSTRUMENTS

The ESP experiment instruments described in this section include:

- o Dual vidicon camera system
- o Accelerometer
- o Stagnation and base region pressure transducers
- o Stagnation and base region temperature transducers
- o Mass spectrometer

The vidicon camera system is used to obtain surface images and the balance of the instruments are used for the atmosphere profile determination.

1.1 ENTRY TELEVISION

1.1.1 Equipment Identification and Usage - This equipment consists of a dual vidicon camera system mounted to the bottom of the CBS impact foot pad and viewing out through a window in the Aeroshell. One camera (CAM 1) provides a high resolution/narrow field of view capability, the other camera (CAM 2) provides wide angular converage at reduced resolution. The units are housed in a common protective container and are pointed parallel to the Capsule roll axis. They cycle alternately and provide continuous imagery from altitudes of 800,000 feet to near touchdown.

Each camera is comprised of a lens, vidicon, shutter/iris control/filter wheel, control electronics, and power supply. These are functionally organized as shown in Figure 1.1-1. The characteristics of these components are summarized below.

1.1.1.1 Lenses - The lenses collect light from the Mars scene and image it on the detectors. CAM 1 employs an F/1.5, 3.2 inch focal length lens and gives an 8° field of view when used with a 0.44 inch detector format. CAM 2 uses an F/1, 0.5 inch focal length lens which provides a field of view of approximately 50°.

1.1.1.2 Vidicons - Slow scan, 200 x 200 line, 6 bit gray scale vidicons with 0.44 inch square sensitive areas are used for both cameras. The sterilizable vidicon with an antimony-sulfide-oxy-sulfide (ASOS) photocathode and electrostatic deflection/focusing is used. This conversion is implemented by systematically scanning an electron beam over the spatial charge pattern stored on the photocathode after exposure to the scene luminance. Following the electron beam readout of the image charge pattern on the photocathode, the photocathode is cleared of residual charge by irradiating it with a lamp. This erasure process supplies a fresh, noise-free

FUNCTIONAL BLOCK DIAGRAM – DESCENT TV

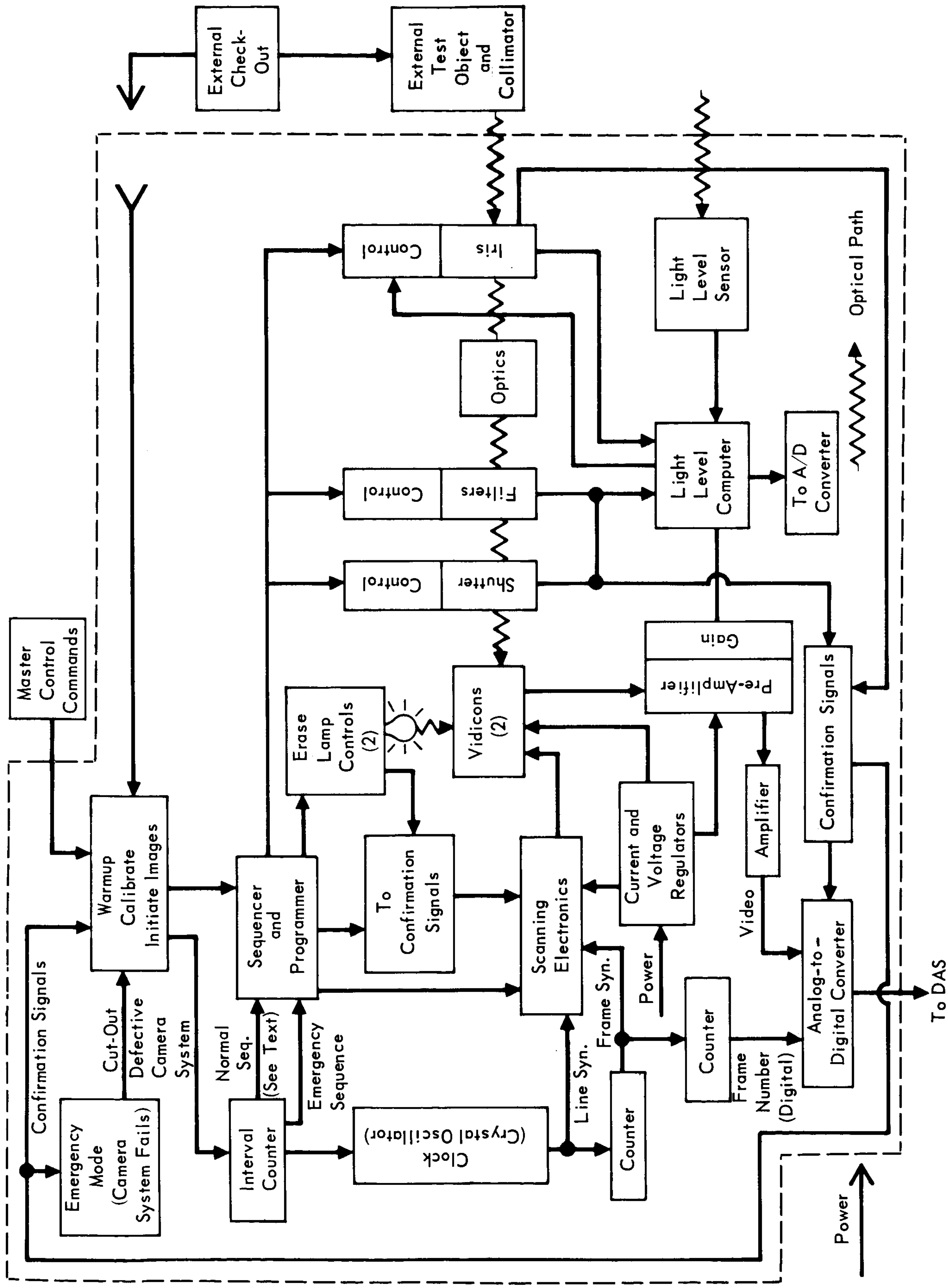


Figure 1.1-1

surface, ready to receive the next optical image.

1.1.1.3 Shutter/Iris Control/Filter Wheel - This unit controls the duration, amplitude, and color of the light which reaches the vidicon photocathode. The shutter/iris portion is a variable sized hole, located directly behind (or as an integral part of) the lens, which can be opened and closed in the few milliseconds it takes to form a charge image on the photocathode of the vidicon. Hole size is set automatically by a small photodiode which senses the intensity of the incident illumination. The filter wheel is located adjacent to the shutter/iris. It contains a series of alternately neutral (clear) and minus blue filters mounted on a rotary wheel. These filters are sequenced into the optical path providing for either a wide spectral band or red enhanced picture.

1.1.1.4 Control Electronics - These elements insure proper command sequencing, operational synchronization, and data handling. They initiate the camera warmup, the imaging cycle, and functional operations such as filter wheel positioning and temperature monitoring. Vidicon processing requires scan/readout/erasure/synchronization and camera-to-camera correlation. Signals must be amplified, sequenced (once every five seconds), and converted to a digital format. An internal sense/shutdown mode is needed for each camera to avoid total system failure in the case of serious problems in one of the units.

1.1.1.5 Power Supply - Regulation and voltage conversion is accomplished here. Vidicons need to develop 1000-2000 volt internal potentials. Electronics and shutter/filter wheel assemblies require stable, thermally independent voltages and currents following the acceptance of the basic ESP 28.5 ± 5 volt power.

1.1.2 Design Requirements and Constraints - Entry heating causes communications blackout and flow field modification during the middle of the trajectory. The shallow descent angles create problems in keeping the landing site location in view. Descent engine failure could limit the time to transmit terminal pictures. Based upon these three fundamental considerations the cameras have to be designed to obtain backup landing site imaging above 500,000 feet (CAM 1), primary site imaging at 100,000 feet (CAM 2), and terminal surface imaging at 4500 feet (CAM 1).

The cameras will be designed to minimize potential communications blackout problems associated with entry into the Mars atmosphere.

Peak heating can cause self-luminous gas emission and ablative deposition on optical windows. This is avoided by designing a non-ablative nose-cap forward of the camera window location on the Aeroshell. High thermal fluxes (temperatures to 700°C) at this location force the use of fused silica window glass and heat reflective optical coatings in order to maintain a workable system. Coatings must be applied on the lenses as well as the insulated container in order to secure good optical focus and stable electrical performance. These provisions are also useful in the terminal phases when descent engine heating becomes a problem.

Landing site identification requires at least 1000 foot ground resolution. To achieve suitable coverage after blackout the optics must be designed with a 50° field of view. These two numbers are compatible with imagery from altitudes below 150,000 feet. Terminal descent rates could exceed 750 feet/second. These values lead to a camera with a 8° field of view when 1 meter ground resolution is required.

Secondary considerations relate to image motion and obscuration. Exposure at 10 milliseconds (established by scene lighting) requires Capsule angular rates to be below 4 degrees/second to avoid image smear. Blocking of the optical path will occur at Aeroshell separation and large oscillations may be experienced here as well as at parachute deployment and descent engine ignition. Cycle rates of at least one picture every 5 seconds are useful here in order to secure correlation and coverage.

1.1.3 Physical Description - Conceptually the camera system is as pictured in Figure 1.1-2.

The dual cameras are located side-by-side to facilitate packaging, thermal protection, and electrical control. Each unit posseses internal optical alignment and viewing is conducted out one end of the package simplifying installation. Thermal protection is provided through an insulated shell having a heat reflective outer skin. Electrical controls, although independent for each camera, are located physically near another end of the container. This provides a short path to the phototubes .

The camera package averages 8 inches in diameter and is slightly over 12 inches long. It looks through a trapezoidal window in the aeroshell roughly 10 inches long, 7 inches wide and 0.25 inches thick.

1.1.4 Operational Description - The objectives emphasize good landing site identification at high to moderate altitudes and detailed pictorial coverage of the surface at low altitude. This is achieved by a continuous recycling of the system, alternating the exposure of first one camera then the other. One of these cycles is summarized in Figure 1.1-3. Note that the two cameras function almost identically. Camera 1 starts the cycle with a 7 millisecond exposure of the Mars scene. This is followed by a 5 second readout period for vidicon 1 in which electrical data is generated at 50,000 bits per second. At this time the iris control is

DESCENT TELEVISION CAMERAS

General Characteristics	
Weight –	
Optics	2 pounds
Vidicons	1.5 pounds
Mechanism	1.5 pounds
Electronics	5 pounds
Structure	4 pounds
	14 pounds
Power –	
Cameras	18 watts
Controls	2
	20 watts
Volume –	<700 cu inches

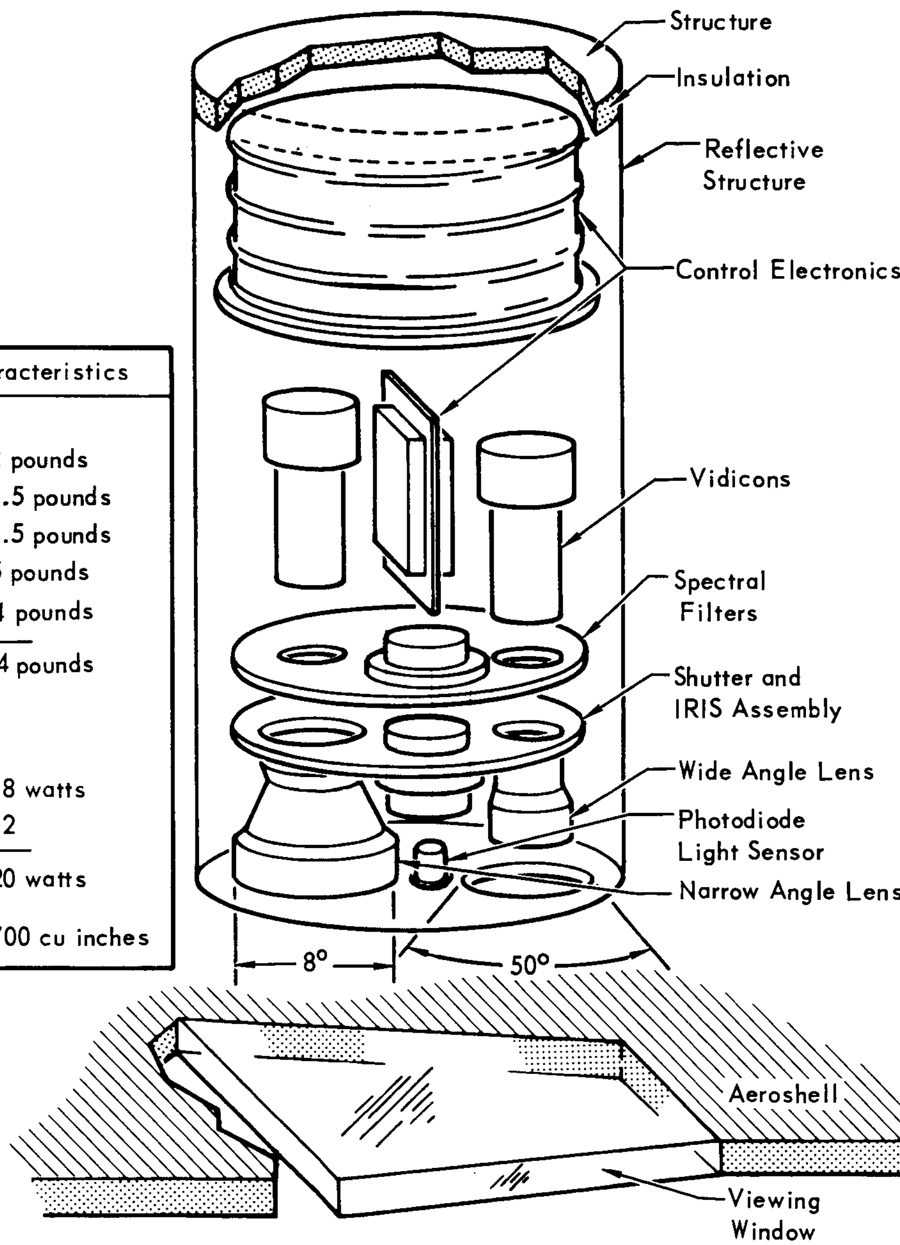


Figure 1.1–2

OPERATIONAL SEQUENCE – DESCENT TV

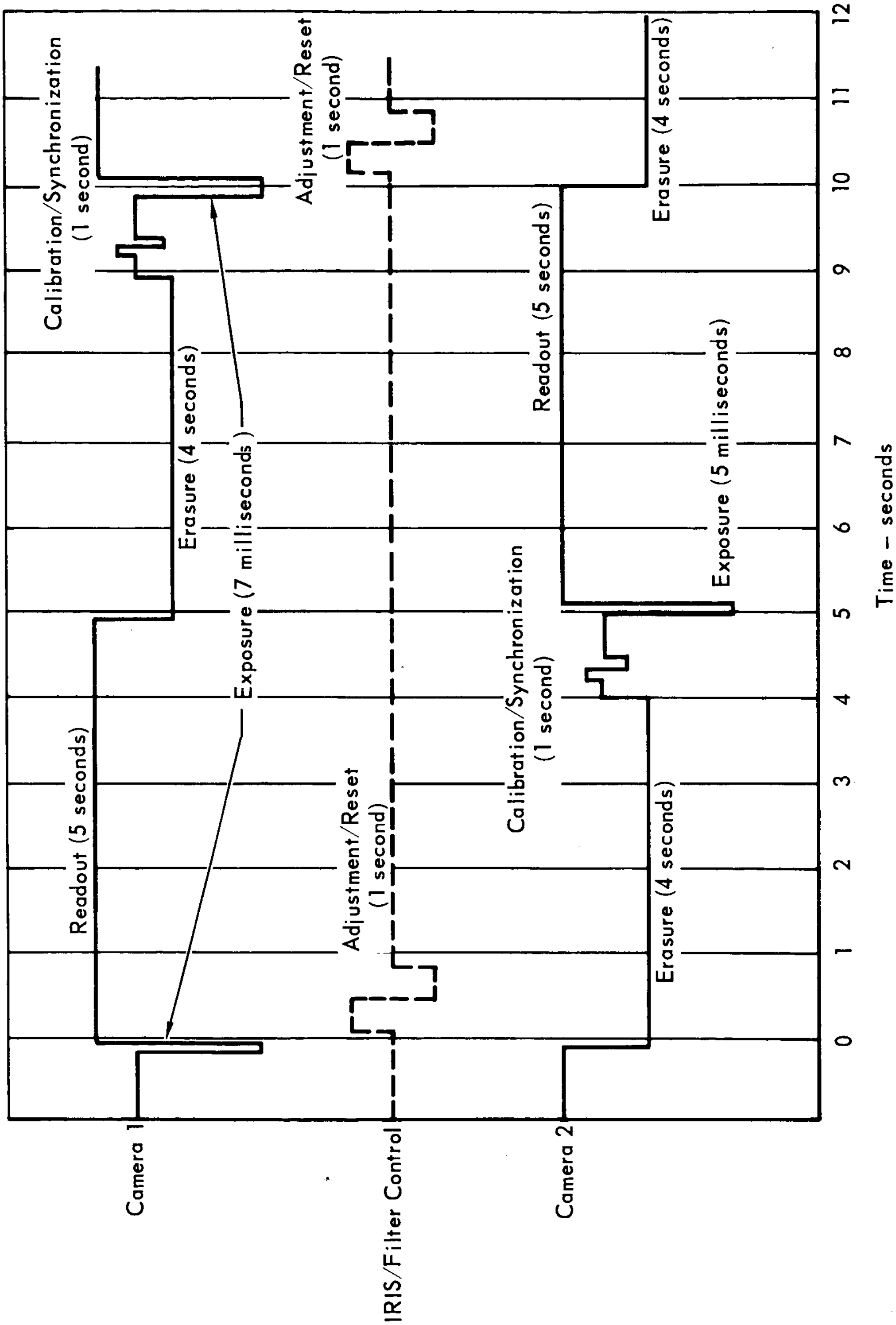


Figure 1.1-3

adjusted to establish the proper tube illuminance based upon the input received from its photodiode sensor. In addition, the filter wheel, initially on a clear setting, is repositioned to introduce a minus blue contrast enhancement filter. This adjustment/reset function is accomplished simultaneously for cameras 1 and 2. The next event is the 5 millisecond exposure of camera 2 followed immediately by its readout. Data handling is established to give a nearly continuous output, i.e., the second readout begins immediately after the first is completed. Upon completion of readout, a four second erasure process is introduced in which vidicon 1 is electro/optically prepared to receive the next exposure. This is followed by a 1 second calibration/synchronization step which ensures proper operation and correlation. The second exposure on camera 1 is then made and its period begins anew. Camera 2 in the meantime completes its readout phase and initiates erasure. Adjustment and reset occur once again giving each camera a new energy input condition.

This process takes place continuously from turn-on at 800,000 feet to destructive termination at 90 feet.

1.1.5 Performance Objectives - Consistant with the overall mission objectives, camera 1 has approximately 0.7 milliradian angular resolving power and an 8° (0.44 inch at the vidicon) field of view. Camera 2 provides a resolution of 40 milliradians at 50°. Exposure times vary from 5 to 10 milliseconds dependent on lighting. The cameras operate in alternating phase so one picture occurs each 5 seconds. Spectral filtering occurs every other frame with a given setting on one camera repeating every twenty seconds.

1.1.6 Interface Definition - The basic interface requirements are outlined in Figure 1.1-4. In addition the following items apply:

Mounting

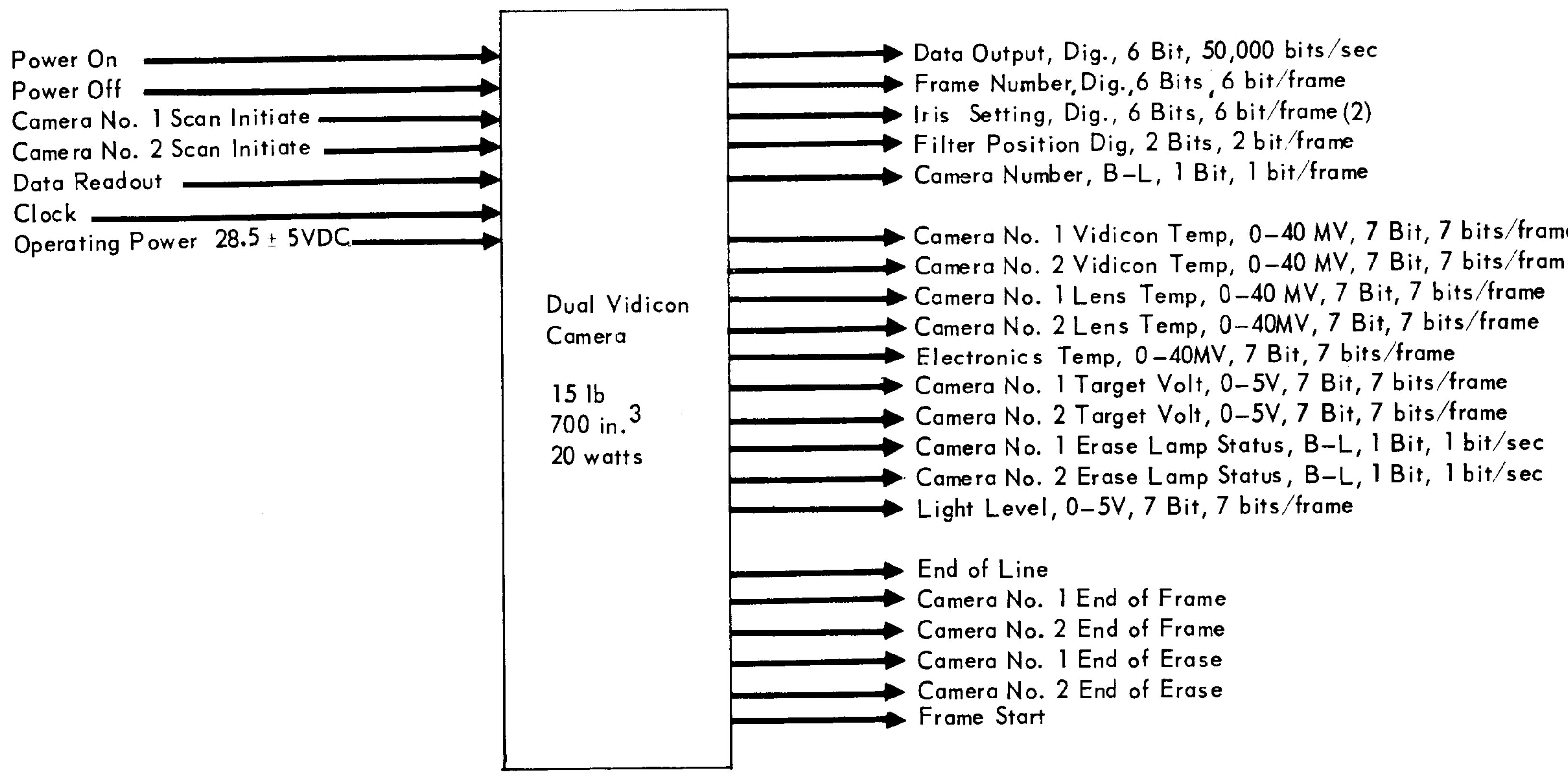
- Positioning on CBS foot pad in alignment with the Aeroshell window.
- Protection of optical surfaces from contamination during sterilization.
- Optical window through sterilization canister to allow optical testing before launch.

Thermal

- Nose-cap design free of ablative contaminates for clear optical viewing during descent.

1.1.7 Reliability and Safety Considerations - The preferred design concept has been developed around a simple utilization concept. There is no dependence on complex mechanisms or detailed programming sequences. If the individual components function (they are all state-of-the-art items), the mission objectives will be

ENTRY TELEVISION INTERFACE DIAGRAM



(B–L) Bilevel
(Dig) Digital

Figure 1.1–4

accomplished. Should some fail, the parallel (two cameras) operational plan ensures at least partial success. Major failure negates only the imaging experiment and does not endanger the rest of the CBS.

1.1.8 Test Requirements - Pre-flight testing consists of two parts. The first is a thorough functional checkout after sterilization in which the cameras undergo complete electrical and mechanical evaluation. Resulting outputs investigated for linearity, noise, sensitivity, and stability. Once the unit is on the launch vehicle a final electro-mechanical check is again conducted to insure normal operation and proper setup. This need by only a one minute operational run. Just prior to start of descent to the Mars surface, this electro-mechanical check should again be performed. Conducting this before release from the Capsule will provide an excellent baseline reference for the system measurements during the actual descent. In this entry phase, critical voltages and temperatures should be monitored on the average of once every 5 seconds. Should these measurements exceed predetermined levels, the camera can be judged inoperative and shutdown if necessary. These commands can be generated automatically within the instrumentation and consist typically of a) stopping the filter wheel in one position, b) allowing the iris to assume one fixed hole size, and c) shutting down the one camera based on shutter or vidicon failure.

1.1.9 Development Requirements - The most difficult camera component to qualify is the vidicon because of the sterilization requirement. The ASOS photocathode unit, however, has been shown to satisfactorily survive the sterilization procedure. The optics, mechanisms and electronics are typical of what has been used on spacecraft for the past seven years; e.g., Mariner, Surveyor, Ranger, Mercury or Gemini. All items discussed in this concept are within the state of the art. The major development effot will center on combining the individual components in a package compatible with the remainder of the capsule.

1.2 ACCELEROMETER

1.2.1 Equipment Identification and Usage - The function of the accelerometer is to obtain tri-axis acceleration measurements during the entry into the Mars atmosphere. The measurements are taken under varying c.g. locations and are used to supply inputs for the atmospheric density and temperature profile reconstruction calculations by the methods described in Part D, Section 2.2 of this volume. The instrument used to make these measurements is a tri-axis servo force balance accelerometer which has dual ranges of 0 to 5g and 0 to 30g in the roll axis and ± 2g in the pitch and yaw.

1.2.2 Design Requirements and Constraints - In order to meet the mission objective of the accelerometer the following design requirements and constraints are imposed:

- The accelerometer must be aligned parallel to the Capsule Bus (CB) entry configuration pitch, roll, and yaw axes.
- Volume must be minimized to enable all seismic masses to be located at the c.g.
- The accelerometer weight and power requirements must meet the design constraints of 2.0 pounds and 4.0 watts, respectively.
- Accuracy must be sufficient to permit atmospheric profile reconstruction per the method described in Volume IV, Part D, Section 2.2 of this volume.
- Range and frequency response must be compatible with the typical acceleration profile presented as Figure 1.2-1.
- The temperature coefficients must be small enough to permit operation in the thermal environment provided by the Thermal Control Subsystems.
- The threshold and resolution of the roll axis system must be compatible with a $\pm 0.1\%$ accuracy requirement imposed on the 0 to 5g range.
- The unit will have capability for electrical g simulation in each axis for check-out purposes.
- The unit must be designed to tolerate all environments induced by the launch, booster separation, interplanetary cruise, capsule separation, and entry through the Mars atmosphere.
- All interfacing subsystem requirements must be met.

1.2.3 Physical Characteristics - The accelerometer used to measure the entry accelerations is a tri-axis, servo force balance accelerometer which is mounted on the CB entry configuration roll-axis slightly ahead of the c.g. The size, weight, and volume requirements, as illustrated in Figure 1.2-2, are all within the constraints specified in the 1973 VOYAGER Capsule Systems Constraints and Requirements Document, Revision 2, 12 June 1967. Figure 1.2-3 presents a nominal configuration drawing for the tri-axis accelerometer.

1.2.4 Operation Description - A functional block diagram of any one axis of the tri-axis accelerometer is presented as Figure 1.2-4. An acceleration input along the sensitive axis of the seismic mass causes it to deflect. The deflection is sensed by the position detector whose output is amplified for use as an input to the forcer. The forcer input from the amplifier is used to generate a restoring force equal in magnitude and opposite in direction to the seismic mass deflection. An output circuit in the forcer senses the signal to the forcer and routes it to

ENTRY ACCELERATION PROFILE

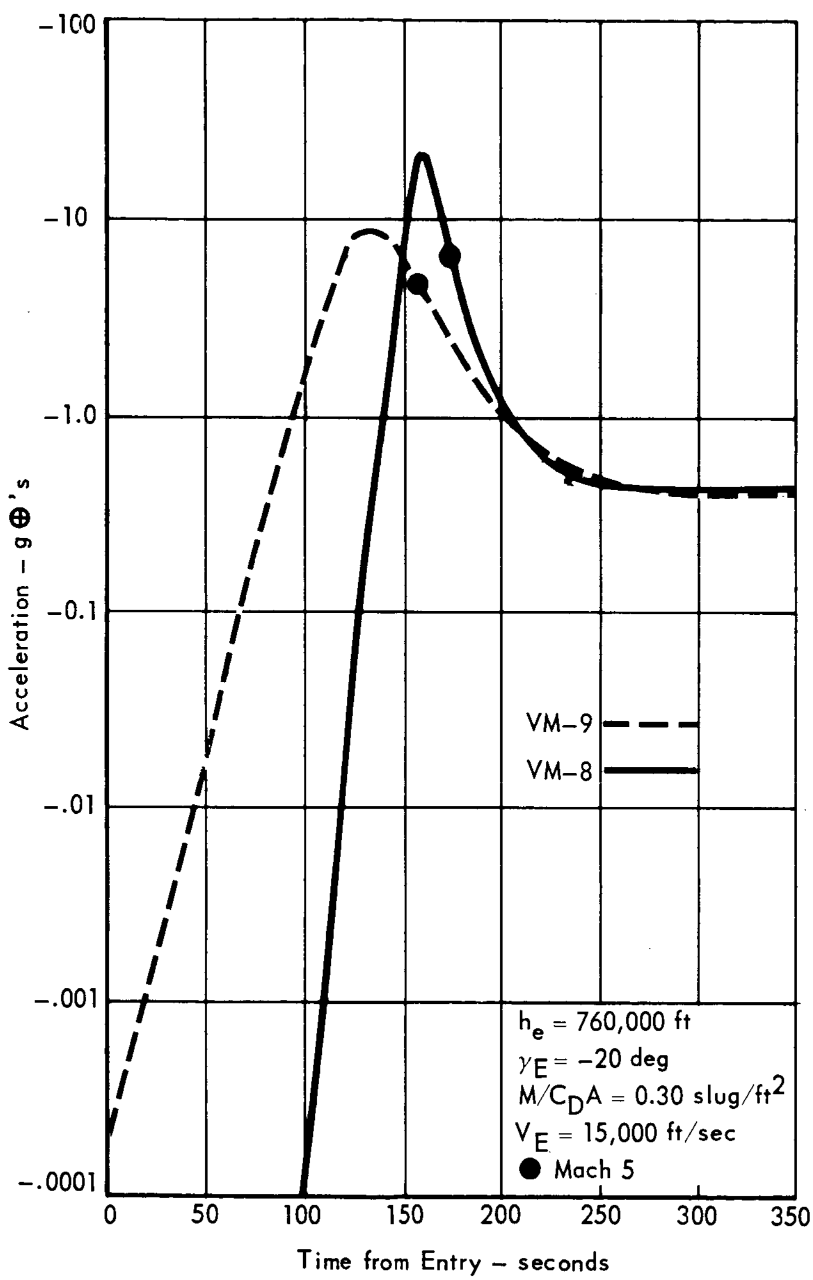


Figure 1.2–1

ESP ACCELEROMETER PHYSICAL CHARACTERISTICS AND CONSTRAINTS

CHARACTERISTIC	PREFERRED INSTRUMENT	CONSTRAINT
Size (in.)	2.75 x 1.75 x 2.0	NA
Weight (lb)	2.0 lbs	2.0 lbs
Volume (in.^{3})	9.625 cu. in.	15 cu. in.

Figure 1.2–2

TYPICAL TRI-AXIS ACCELEROMETER CONFIGURATION DRAWING

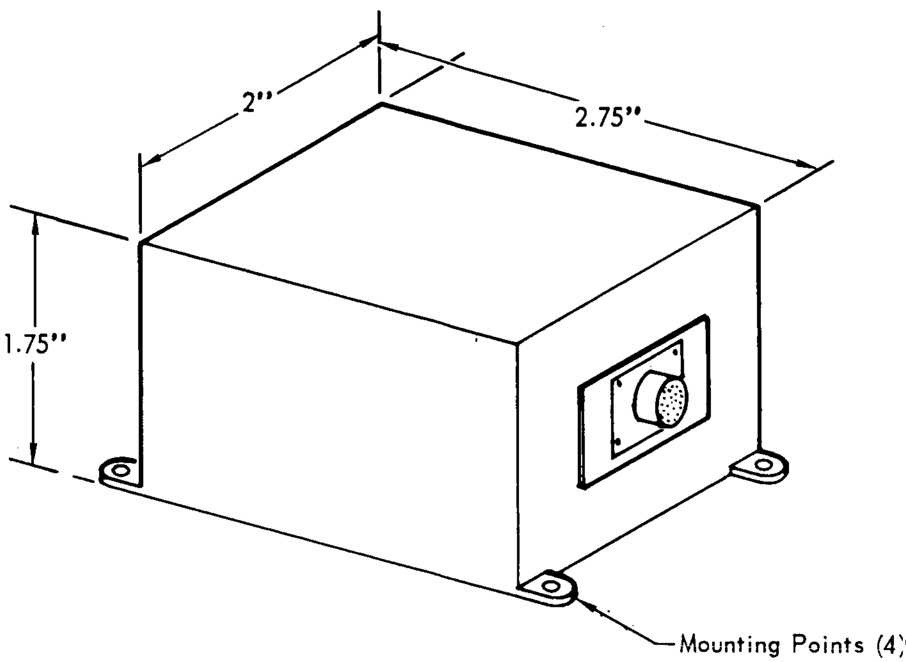


Figure 1.2–3

TYPICAL SERVO FORCE BALANCE ACCELEROMETER FUNCTIONAL BLOCK DIAGRAM

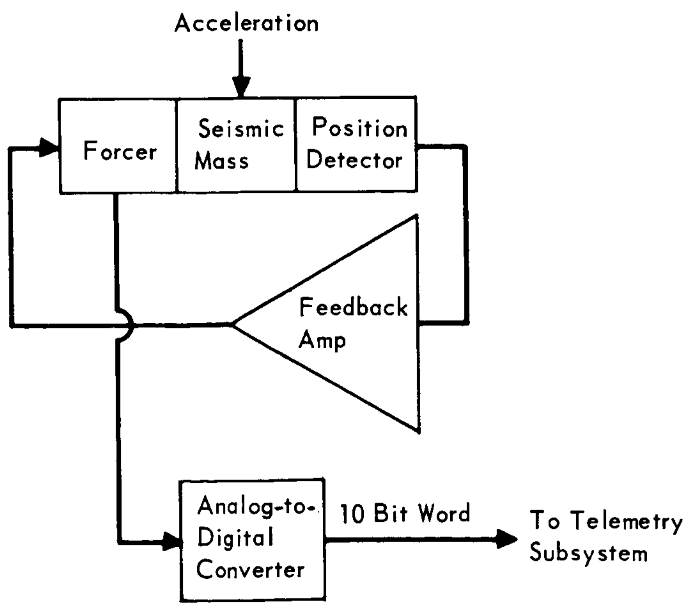


Figure 1.2–4

the analog-to-digital converter in the proper form. The analog-to-digital converter operates on this signal to produce a 10 bit digital output word which is a binary expression of the analog input to the analog-to-digital converter. This function occurs simultaneously in each of the three mutually perpendicular axes to provide the measurements of the tri-axis acceleration.

The tri-axis accelerometer is put into operation along with the other ESP instruments a nominal 300 seconds prior to the entry altitude of 800,000 feet. The instrument continues to operate until the ESP operation is terminated. Data from each of three axes and the range indication for the roll axis is sampled at a rate of 2 samples per second.

1.2.5 Performance Objectives - The accuracy of the accelerometer is ±0.1% of full scale acceleration under any of the environments encountered during the mission. The threshold of the instrument is less than 0.1% of full scale in all axes, and instrument resolution is less than 0.1% of full scale in all three axes. Cross axis sensitivity is kept to a minimum to ensure that roll axis accelerations are only minimally sensed on the transverse axes and vice versa. Figure 1.2-5 presents a typical list of characteristics for this accelerometer.

1.2.6 Interface Definition - Mounting, power, alignment, telemetry, and possibly thermal control interfaces must be compatible for this instrument.

The accelerometer will be mounted ahead of the c.g. of the CB on the roll axis of the CB in its entry configuration. The mounting attachment must provide the means for aligning the accelerometer parallel to the CB entry configuration pitch, roll, and yaw axes. An additional mounting requirement is imposed by the necessity of routing a wire bundle between the accelerometer and the basic ESP which is located at the aft end of the CB.

The accelerometer will use the 28.5 ±5 VDC power level available from the power system and will perform internally any required voltage level changing or regulation.

The Telemetry subsystem interface consists of the three 10 bit digital acceleration outputs plus the 0 or 28.5 VDC roll range indication output from the accelerometer.

The accelerometer design may dictate a requirement for Thermal Control subsystem support, if the accelerometer thermal environment spread exceeds the design thermal limits. Figure 1.2-6 presents the accelerometer interface diagram.

1.2.7 Reliability and Safety Considerations - The reliability factor for obtaining entry accelerations has been improved by the incorporation of functional redundancy

TYPICAL ESP ACCELEROMETER CHARACTERISTICS

Range	Roll Axis: 0 to 5 g and 0 to 30 g Pitch, Yaw Axis: ± 2 g
Output Signals	Acceleration (3) – 10 bit digital words Roll Range: 0 or 28.5 Vdc
Output Rate	Acceleration (3) – 20 bps Roll Range – 2 sps
Input Voltage	28.5 ± 5 Vdc
Input Power	4.0 Watts Nominal
No. of Measurements per Data Channel	2×10^3 Over a Nominal 1000 sec Operating Period
Accuracy	± 0.1% of Full Scale Under Any Mission Induced Environment
Filters	Flat from dc to 1 cps with an 18 dB per Octave Roll-off
Warm-Up Time	60 Seconds Maximum
Threshold	Less than 0.1% of full scale in all axes

Figure 1.2–5

ESP ACCELEROMETER INTERFACE BLOCK DIAGRAM

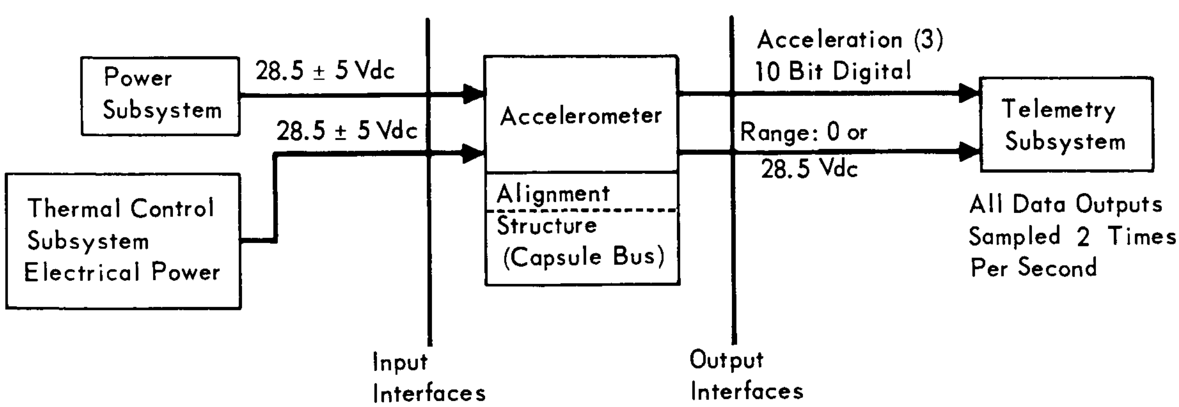


Figure 1.2–6

in the CB subsystem. The IMU subsystem in the CB contains a roll axis accelerometer and three axis gyros which telemeter CB attitude rates, attitude errors, and z-axis accelerations. The Radar Altimeter will provide altitude readings from a nominal 200,000 feet to 18,000 feet.

There are no unique safety considerations required for this instruments.

1.2.8 Test Requirements - After the LV/PV mate, the accelerometer will be given an end-to-end check, which includes an electronic simulation of acceleration levels, to verify launch readiness.

Pre-separation checkout requirements include electrical simulation of accelerations in all three axes and readout of those accelerations.

1.2.9 Development Requirements - The techniques and materials required to implement this instrument are within the state-of-the-art. Some minor development work is required to slightly modify these instruments for use as tri-axis rather than three signel axis units, but the problem is mainly one of packaging.

1.3 PRESSURE TRANSDUCERS

1.3.1 Equipment Identification and Usage - The function of the ESP pressure transducers is to obtain stagnation and base region pressure measurements during the CB entry into the Mars atmosphere. The data generated by these sensors is used to determine free stream pressure versus altitude relations, to provide inputs for other atmospheric property profile determinations, and to supply information for entry trajectory calculations. The instrument selected to obtain these measurements is a variable-capacitance transducer in which a pressure difference between two sides of a diaphragm or membrane causes it to deflect, with the deflection magnitude detected by capacitor plates rigidly mounted near the deflecting member. The capacitance difference between the two plate/deflecting member capacitors provides an output which is a nearly linear function of applied pressure.

1.3.2 Design Requirements and Constraints - The design of the transducers is affected by constraints associated with interfacing subsystems, mission objectives, pre-launch checkout requirements, and mission-induced environments.

These pressure transducers interface with the ESP power, structural, telemetry, and possibly the thermal control subsystems. Design constraints due to interfacing subsystems are:

- o Voltage and power levels of the power subsystem.
- o Mounting points, clearance, and atmospheric access requirements on the structural subsystem.
- o Telemetry subsystem input signal levels and impedance characteristics.

o Possible necessity of thermal protection of the diaphragms and electronics. The mission objective is to obtain pressure measurements during entry for use in atmospheric profile and trajectory characteristics reconstructions. This objective imposes accuracy and response time design requirements on the instruments and on their operating periods during entry and descent.

Pre-launch checkout procedures are, in part, conducted under Earth ambient conditions, thereby imposing a requirement that the transducers be capable of tolerating one atmosphere pressure in the operating mode without experiencing permanent changes in performance characteristics. In order to perform successfully after the sterilization cycle, the transducers must be able to withstand non-operating pressures of up to 23.0 psia and long-term high temperature exposure.

In addition to the above requirements, the transducers must be capable of tolerating all environments induced by the launch, booster separation, interplanetary cruise, separation from the spacecraft, and entry through the Mars atmosphere.

1.3.3 Physical Characteristics - The pressure transducers used to measure the base and stagnation pressures are housed in a 2 inch diameter cylinder which is approximately two inches long. The sensing element is a free-edged diaphragm mounted between two rigidly attached capacitor plates. Figure 1.3-1 illustrates the internal configuration of a typical variable capacitance transducer sensing element. This unit coupled with the input and output electronic circuitry comprises the pressure transducer. Figure 1.3-2 contains a summary of the physical characteristics of the pressure transducers, and Figure 1.3-3 presents the nominal configuration for the unit including the pressure inlet.

1.3.4 Operation Description - The operation of the transducers in based on pressure induced deflections, corresponding capacitance changes, and subsequent production of a DC output voltage proportional to the capacitance difference. The capacitance difference is used as the output function to obtain output linearity. A typical output circuit which will provide a DC output proportional to applied pressure is shown in Figure 1.3-4. A positive oscillator output causes a positive current to be fed to the filter capacitor and load resistor from the upper capacitor while the lower capacitor current is routed to ground. A negative oscillator results in a negative current from the lower capacitor being fed to the filter capacitor and load resistor and the current through the upper circuit being routed to the ground. The resultant output is a voltage proportional to the capacitance difference. A functional block diagram is shown as Figure 1.3-5 and the transducer characteristics are tabulated as Figure 1.3-6.

CROSS SECTION VIEW OF TYPICAL VARIABLE CAPACITANCE PRESSURE TRANSDUCER SENSING ELEMENT

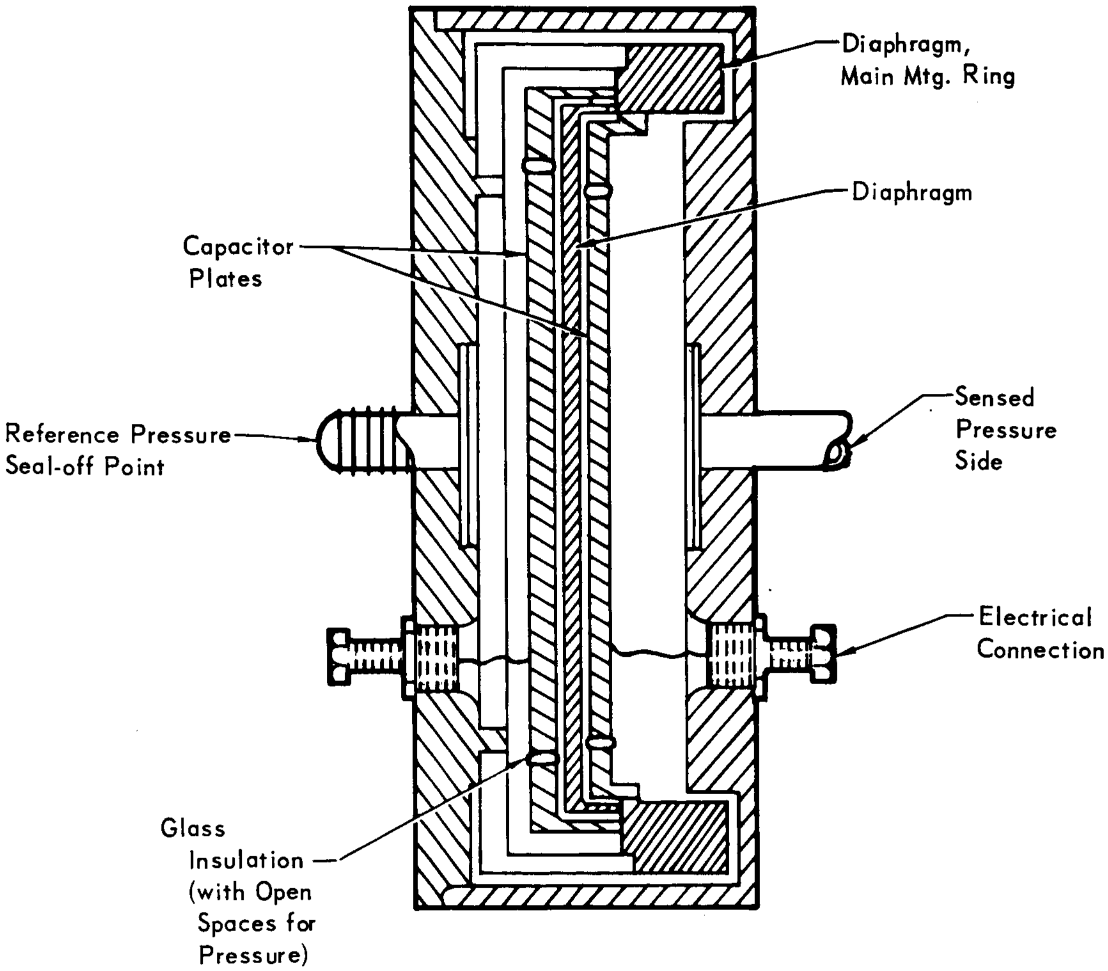


Figure 1.3–1

PHYSICAL CHARACTERISTICS OF THE ESP PRESSURE TRANSDUCERS

Size	2 inches in diameter x 2 inches long
Weight	1.0 lb each
Volume	6.3 cubic inches
Case Material	304 Stainless Steel

Figure 1.3–2

TYPICAL PRESSURE TRANSDUCER EXTERNAL CONFIGURATION

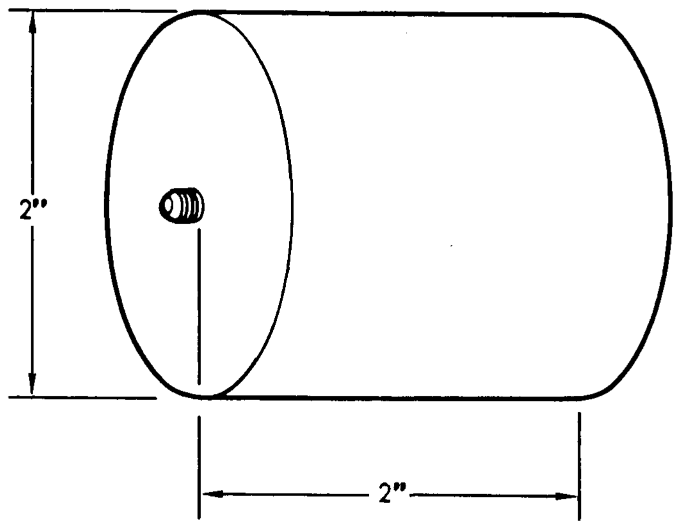
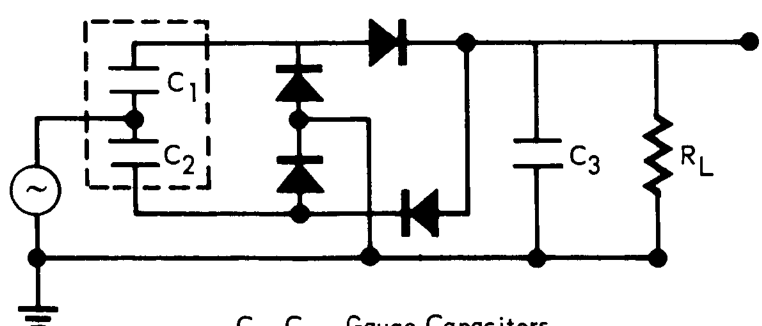


Figure 1.3-3

TYPICAL ATMOSPHERIC PRESSURE TRANSDUCER OUTPUT CIRCUIT



C_1, C_2 = Gauge Capacitors

C_3 = Filter Capacitor

R_L = Load Resistor

Figure 1.3-4

TYPICAL PRESSURE TRANSDUCER FUNCTIONAL BLOCK DIAGRAM

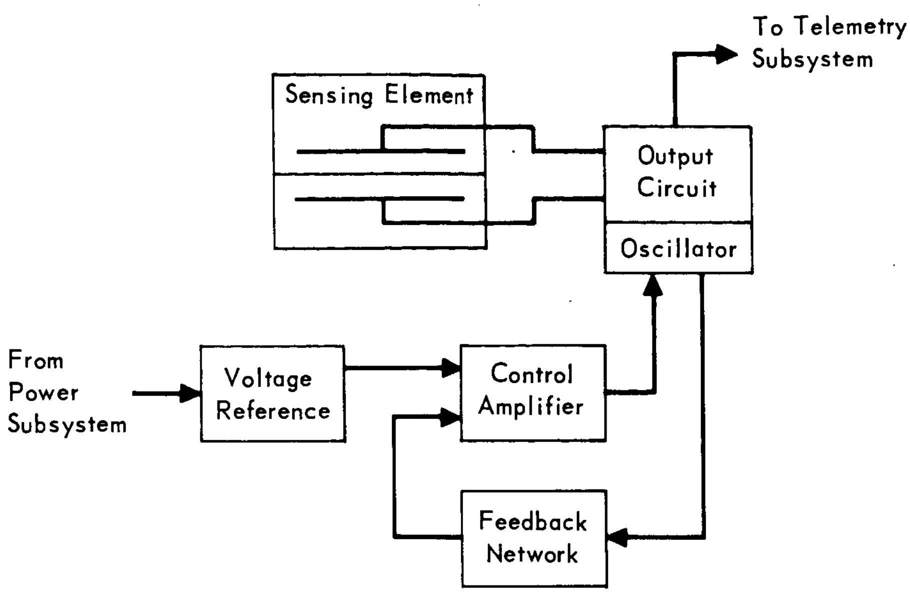


Figure 1.3–5

ESP PRESSURE TRANSDUCER CHARACTERISTICS

CHARACTERISTICS	STAGNATION TRANSDUCER	BASE TRANSDUCER
Range	0–3 psia	0–.75 psia
Proof Pressure	23 psia	23 psia
Burst Pressure	30 psia	30 psia
Warm-up Time	60 sec	60 sec
Output Voltage	0 to 5 vdc	0 to 5 vdc
Input Voltage	28.5 ± 5 vdc	28.5 ± 5 vdc
Input Power	1.4 watts	1.4 watts
Time Constant* at 0.1 psi	15 msec	15 msec
Accuracy	±0.5% of f.s.	±0.5% of f.s.

*This number represents the time constant for the sensor with no porting tube attached.

Figure 1.3–6

The stagnation pressure transducer will be put into operation a nominal 300 seconds prior to reaching the entry altitude of 800,000 feet. The transducer will be sampled by the telemetry subsystem at a rate of one sample per second during its entire operating period which extends from turn-on until Aeroshell separation at a nominal 18,000 feet altitude. Meaningful data will be obtained when the stagnation pressure reaches a point high enough to generate a non-zero output in the transducer and will continue until Aeroshell separation. Since the stagnation transducer separates with the Aeroshell, the wire bundle will be separated just prior to the Aeroshell separation and the load on the Power Subsystem and the output from the transducer will no longer be present.

The base region pressure transducer is activated along with the stagnation pressure transducer at 300 seconds prior to entry and remains in operation until 600 seconds after touchdown. The base region transducer is also sampled at a rate of once per second. Data will be obtained from the base region transducer throughout entry and descent and until 120 seconds after touchdown on the Mars surface.

1.3.5 Performance Objectives - The accuracy of the pressure transducers is $\pm$0.5% of full scale in the environment produced by entry into the Mars atmosphere. Resolution of the capacitance function is continuous and the limiting factor in determining pressure resolution is the word length of the analog-to-digital converter in the Telemetry Subsystem. The dual capacitor readout technique is used because the single capacitance variation is nonlinear but the increase in one and the corresponding decrease in the other result in a capacitance ratio which is linear within $\pm$0.5% of full scale. The time constant must be kept small to obtain accurate readings at a rate of one per second. The unit response time is less than 50 milliseconds for the lowest pressure of interest decreasing to 15-20 milliseconds at 0.1 psia and to 5 milliseconds at 1 Earth atmosphere.

The construction of this transducer must minimize the effects of acceleration induced diaphragm deflection. The acceleration sensitivity is equal to or less than 0.02% of full scale per g in the most sensitive axis.

Calibration stability over the range of environment and duration in each environment precludes the necessity of in-flight calibration or monitoring.

1.3.6 Interface Definition - Mounting, power, telemetry, and possibly thermal control interfaces must be compatible for these instruments as shown in Figure 1.3-7 and 1.3-8 which present the stagnation and base pressure interface diagrams respectively.

The stagnation transducer will be mounted on the stagnation temperature

STAGNATION PRESSURE TRANSDUCER INTERFACE BLOCK DIAGRAM

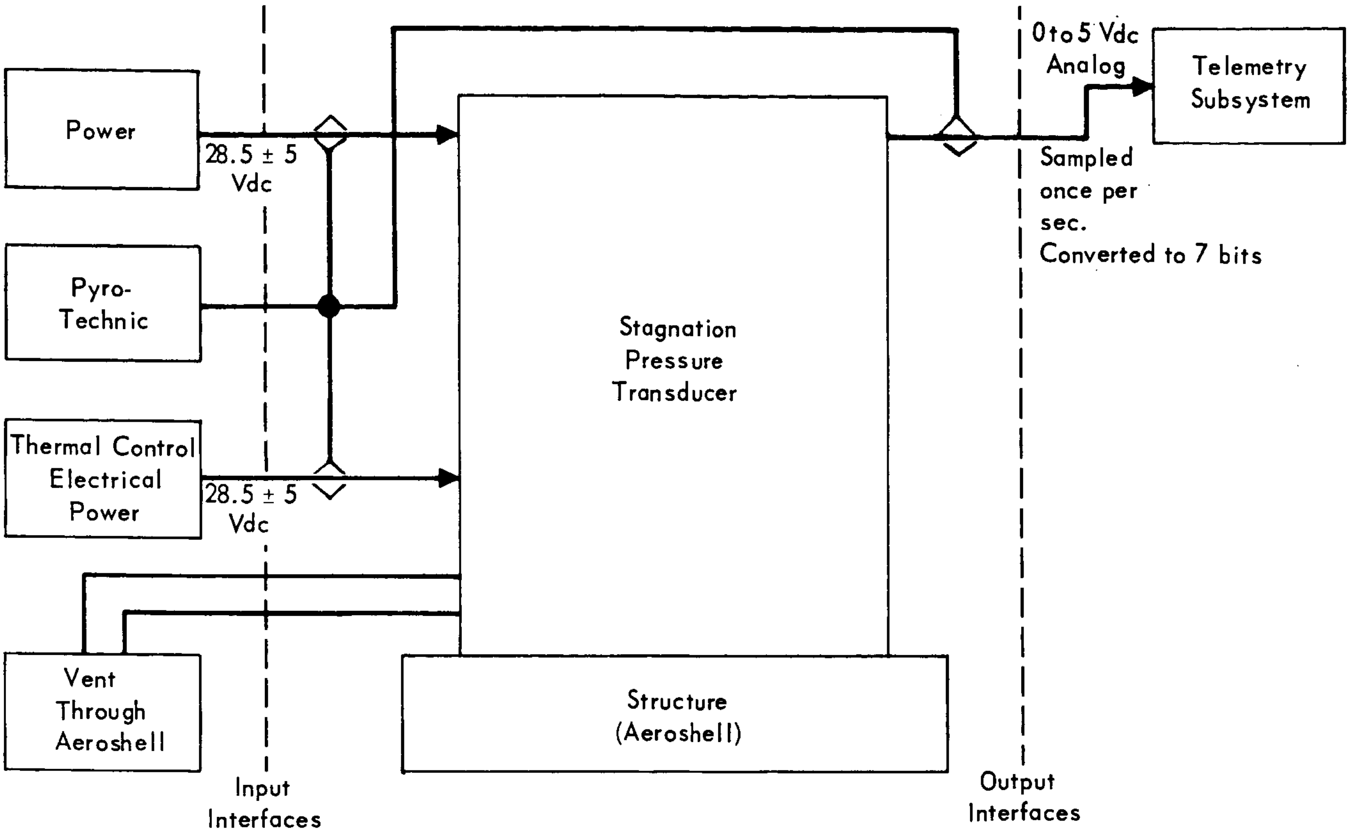


Figure 1.3-7

BASE PRESSURE INTERFACE BLOCK DIAGRAM

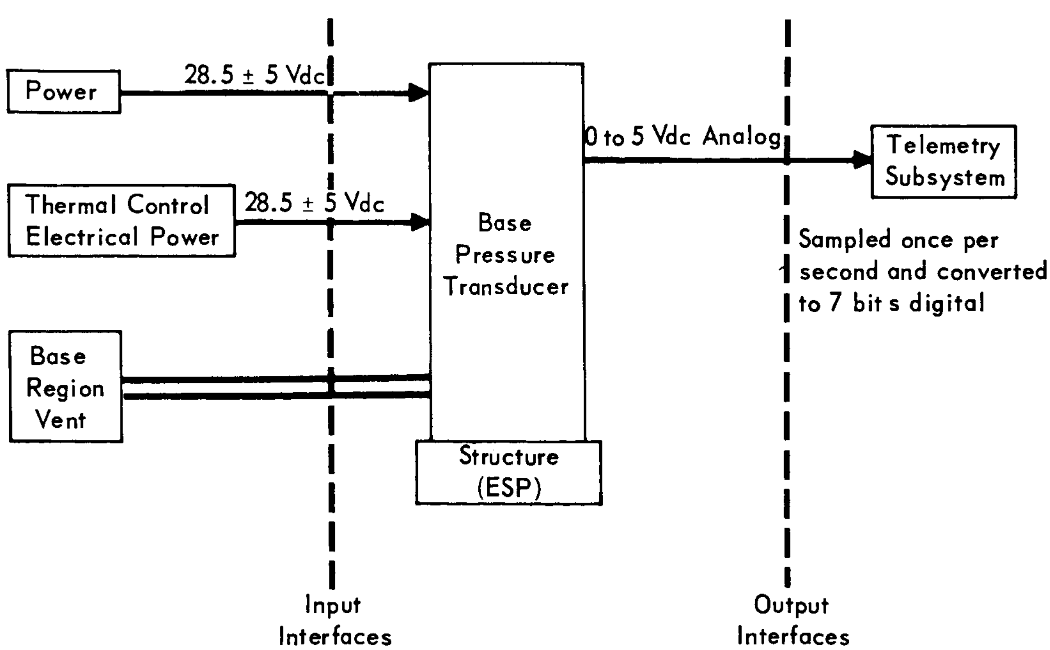


Figure 1.3-8

assembly which is attached to the Aeroshell. Access to the atmosphere is via a concentric cylinder around the outer radiation shield on the temperature sensor.

The base region sensor is mounted on the aft end of the ESP with direct access to the atmosphere in the base region.

Each unit will use 1.4 watts of the 28.5 ± 5 Vdc power level available from the Power Subsystem and each will provide a 0 to 5 Vdc output to the Telemetry Subsystem.

The thermal environment induced on the transducers may require Thermal Control Subsystem interfaces to satisfy two transducer design requirements:

- o Thermal insulation mounting to keep the stagnation pressure transducer diaphragm temperature at an acceptable level.
- o Electronics temperature maintenance within design limits for both units.

1.3.7 Reliability and Safety Considerations - In order to ensure reliable performance by the pressure transducers, a post-sterilization calibration will be performed to verify that the reference cell has remained sealed during exposure to the thermal environment of the sterilization cycle.

Safety considerations require a burst pressure of 30 psi or more.

1.3.8 Test Requirements -

Pre Flight: After the LV/PV mate, an end-to-end check will verify pressure transducer integrity with the overall system.

In Flight: No in-flight monitoring or checkout is required other than simple voltage and continuity checks.

1.3.9 Development Requirements - The techniques and construction methods required to implement the pressure transducers are well within the state-of-the-art. However, modest development efforts will be required in the following areas:

- o Upgrading of welding techniques to ensure that the thermal sterilization cycles do not cause the reference cell seal to be broken.
- o Determination of maximum temperature that the diaphragm will withstand without significant performance degradation.
- o Quantization of the sterilization-induced temporary calibration shifts.
- o Determination of base and stagnation pressure coefficients as functions of Mach number and angle-of-attack.

1.4 TEMPERATURE TRANSDUCERS

1.4.1 Equipment Identification and Usage - The function of the temperature transducers is to obtain measurements which can be related to free stream temperatures for use in determining atmospheric density and temperature profiles. The most useful temperature measurement data will be obtained from altitudes corresponding to 30 seconds after peak stagnation pressure to a nominal 120 seconds after touchdown. Two total temperature transducers will be used for these measurements - one in the stagnation region which is used from 30 seconds after peak stagnation pressure occurrence until just prior to Aeroshell separation, and one in the base region of the CB which is used from 30 seconds after peak stagnation pressure occurrence until a nominal 120 seconds after touchdown. The total temperature transducers are platinum resistance elements with a resistance bridge output circuit mounted in an aerodynamically desirable shaped housing.

1.4.2 Design Requirements and Constraints - The design of these instruments is affected by constraints with interfacing subsystems, mission objectives, and mission operation considerations.

The temperature transducers interface with ESP power, structural, and telemetry subsystems. Voltage and power levels of the power subsystem, separation of the stagnation temperature transducer wire bundle, and telemetry subsystem input signal levels and impedance characteristics all constitute interface constraints. The stagnation temperature sensor is mounted on the Aeroshell, thus establishing a structural interface with the CB.

The mission objective is to obtain stagnation and base temperature measurements subsequent to the peak stagnation pressure portion of the entry, for use in atmospheric temperature and density profile reconstruction. This objective imposes accuracy and response time design requirements on the instruments. In addition, the stagnation temperature sensor inlet must be able to accept gas flows up to 20^o off centerline due to possible variations in the CB angle-of-attack. The sensing element in the stagnation transducer must be able to withstand the peak heat loads to which it will be exposed during entry. The base region transducer must be shielded from solar radiation during its measurement period.

1.4.3 Physical Characteristics - The temperature transducers used to measure the stagnation and base temperatures are total temperature sensors.

The stagnation transducer sensing element is surrounded by three concentric radiation shields. The mounting of the sensor in the housing provides sensor protection from conducted heat. Attached to the rear of the housing is the vent tube which is routed through the CB to the base.

The base region transducer is a modified total temperature sensor which accepts gas flow from any direction. This unit does not require aerodynamic design to maximize the recovery coefficient since the base region local flow velocity is Mach 0.3 or less during the measurement period. However, a non-heat conducting sensor mounting is used and protection from solar radiation is provided. Figure 1.4-1 illustrates the internal configuration of a typical total temperature sensor. Figure 1.4-2 summarizes the physical characteristics of the stagnation and base region total temperature transducers. Figure 1.4-3 presents an external envelope configuration drawing of a typical total temperature transducer.

1.4.4 Operation Description - The operation of the transducers is based upon the knowledge of the resistance versus temperature relationship for platinum. The transducers use annealed platinum sensing elements and an unbalanced resistance bridge output circuit. A functional block diagram is shown in Figure 1.4-4. Ideally, a total temperature measurement for a gas in high velocity flow is obtained when the gas is brought to rest or nearly so without the addition or removal of any heat. This total temperature is related to the free stream temperature of the gas by the relation

$$T_o = T\left[1 + \frac{(\gamma - 1)\,M^2}{2}\right]$$

where: T_o = total temperature in ^oK

T = free stream temperature in ^oK

γ = ratio of specific heats

M = Mach number

The total temperature sensor does not measure the true total temperature but rather senses a recovery temperature which is slightly less than T_o primarily due to conduction of heat from the sensing element to the surrounding structure. Thus the sensed temperature is related to free stream temperature by the relation

$$T_R = T\left[1 + \frac{r(\gamma - 1)\,M^2}{2}\right]$$

where: T_R = recovery or sensed temperature in ^oK

r = recovery factor

This recovery factor is a function of Mach number which decreases with decreasing Mach number in a non-linear manner. Figure 1.4-5 lists the transducers' characteristics.

INTERNAL CONFIGURATION OF A TOTAL TEMPERATURE TRANSDUCER

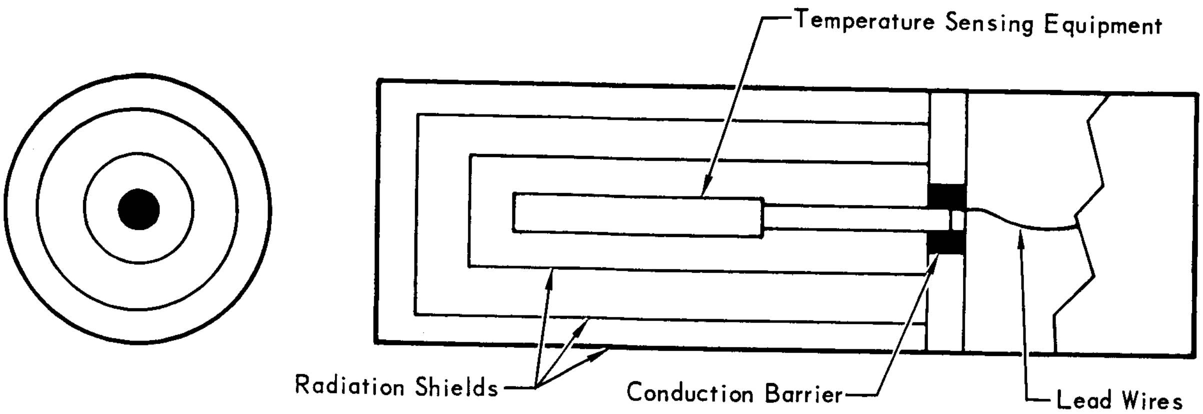


Figure 1.4-1

PHYSICAL CHARACTERISTICS OF THE ESP TOTAL TEMPERATURE TRANSDUCERS

Size	1.0 in. dia x 1.90 in. long
Weight	0.5 lb
Volume (Excluding vent tube for stagnation assembly)	1.7 cu in.

Figure 1.4-2

CONFIGURATION OF ESP TOTAL TEMPERATURE TRANSDUCERS

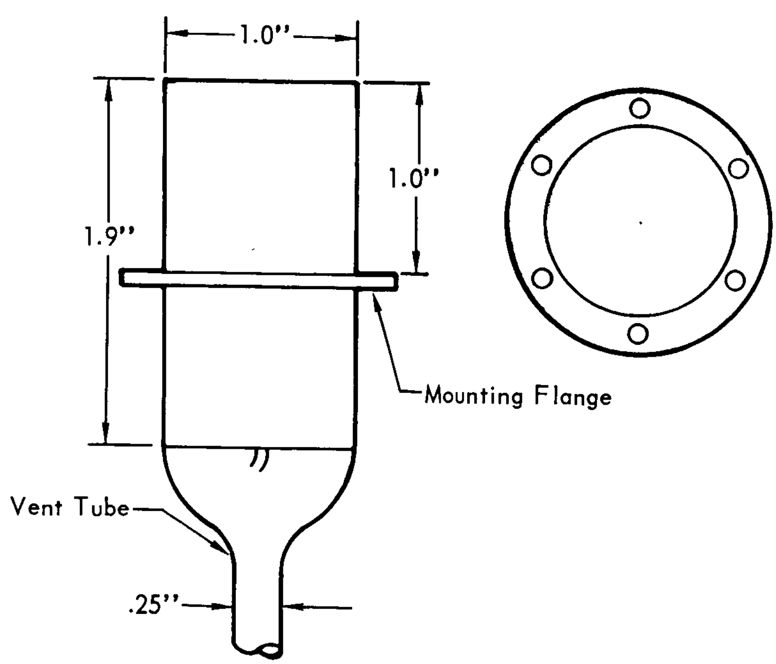


Figure 1.4-3

TEMPERATURE TRANSDUCER FUNCTIONAL BLOCK DIAGRAM

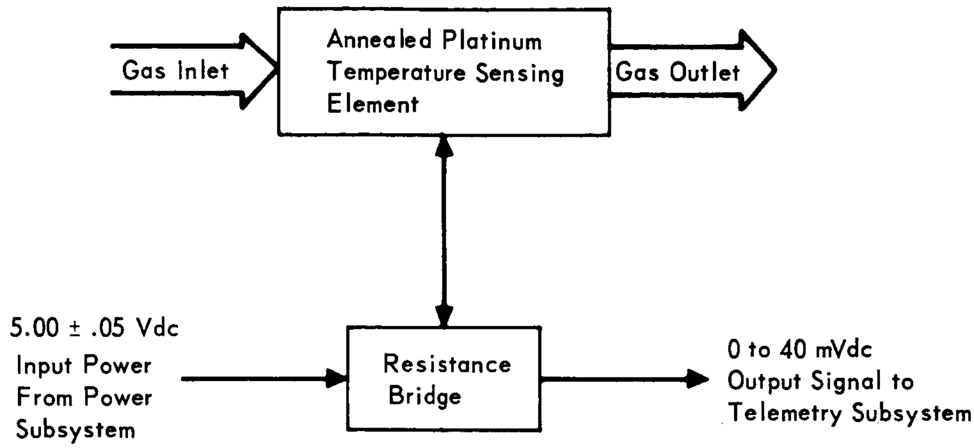


Figure 1.4–4

ESP TEMPERATURE TRANSDUCER CHARACTERISTICS

CHARACTERISTIC	STAGNATION TRANSDUCER	BASE TRANSDUCER
Range	150° to 1200°K	150° to 330°K
Type	Total Temperature Sensor with Platinum Resistance Sensing Element	Total Temperature Sensor with Platinum Resistance Sensing Element
Input Voltage	5.00 ± .05 Vdc	5.00 ± .05 Vdc
Output Voltage	0 to 40 mVdc, double ended	0 to 40 mVdc, double ended
Accuracy	± 1% of full scale	± 1% of full scale
Installed Location	At stagnation point on Aeroshell for 0° angle-of-attack	At aft end of ESP with access to base region
Number of Measurements	1000 Over a 1000 second operating period.	1000 Over a 1000 second operating period.

Figure 1.4–5

The stagnation temperature sensor is put into peration along with the other ESP science instruments a nominal 300 seconds prior to reaching the entry altitude of 800,000 feet. This transducer is sampled by the ESP telemetry subsystem at a rate of one sample per second from turn-on until Aeroshell separation at a nominal altitude of 18,000 feet. Meaningful stagnation temperature measurements will be obtained only after the stagnation temperature outlet tube is opened (about 30 seconds after peak stagnation pressure is sensed). Stagnation temperature measurements will continue until just prior to Aeroshell separation. Since the temperature transducer is mounted to the Aeroshell, its wire bundle must be separated just prior to Aeroshell separation.

The base region temperature transducer is activated along with the stagnation temperature transducer but its operation extends beyond that of the stagnation transducer. This transducer is also sampled at a rate of one sample per second but its operating period extends from 300 seconds prior to entry to a nominal 600 seconds after touchdown. The region during which meaningful data will begin to be obtained during entry or descent will be determined by tests simulating both the CB entry and terminal descent configurations. Although the base region temperature sensor will remain powered up until 600 seconds after touchdown, temperature data will not be transmitted for the entire 600 seconds.

1.4.5 Performance Characteristics - Both transducers will measure recovery temperature to an accuracy of $\pm 1\%$ of full scale. The recovery error for the transducer is a function of Mach number in a manner shown in Figure 1.4-6. Angle-of-attack excursions of 15° will have no noticeable effect on sensed temperatures for Mach numbers above 4.0 and no noticeable effect for 20° angle-of-attack excursions above Mach 3.0. The time constant of the instruments is less than 0.5 seconds over the range of meaningful data output. Sensor calibration stability is sufficient to preclude inflight monitoring or pre-separation checkout.

1.4.6 Interface Definition - Mounting, power, pyrotechnic, and telemetry interfaces must be compatible for these instruments.

The stagnation transducer is attached to the inside of the Aeroshell with the entrance port protruding through the Aeroshell to provide atmospheric access. The vent tube is routed through the CB from the base of the stagnation transducer to the base of the CB. The vent tube is routed by the mass spectrometer to allow a molecular leak attachment which will provide atmospheric samples for mass spectrometer analysis. The base region transducer is mounted on the aft end of the basic ESP with direct access to the base region.

RECOVERY ERROR VS MACH NUMBER BAND FOR TOTAL TEMPERATURE TRANSUDCER

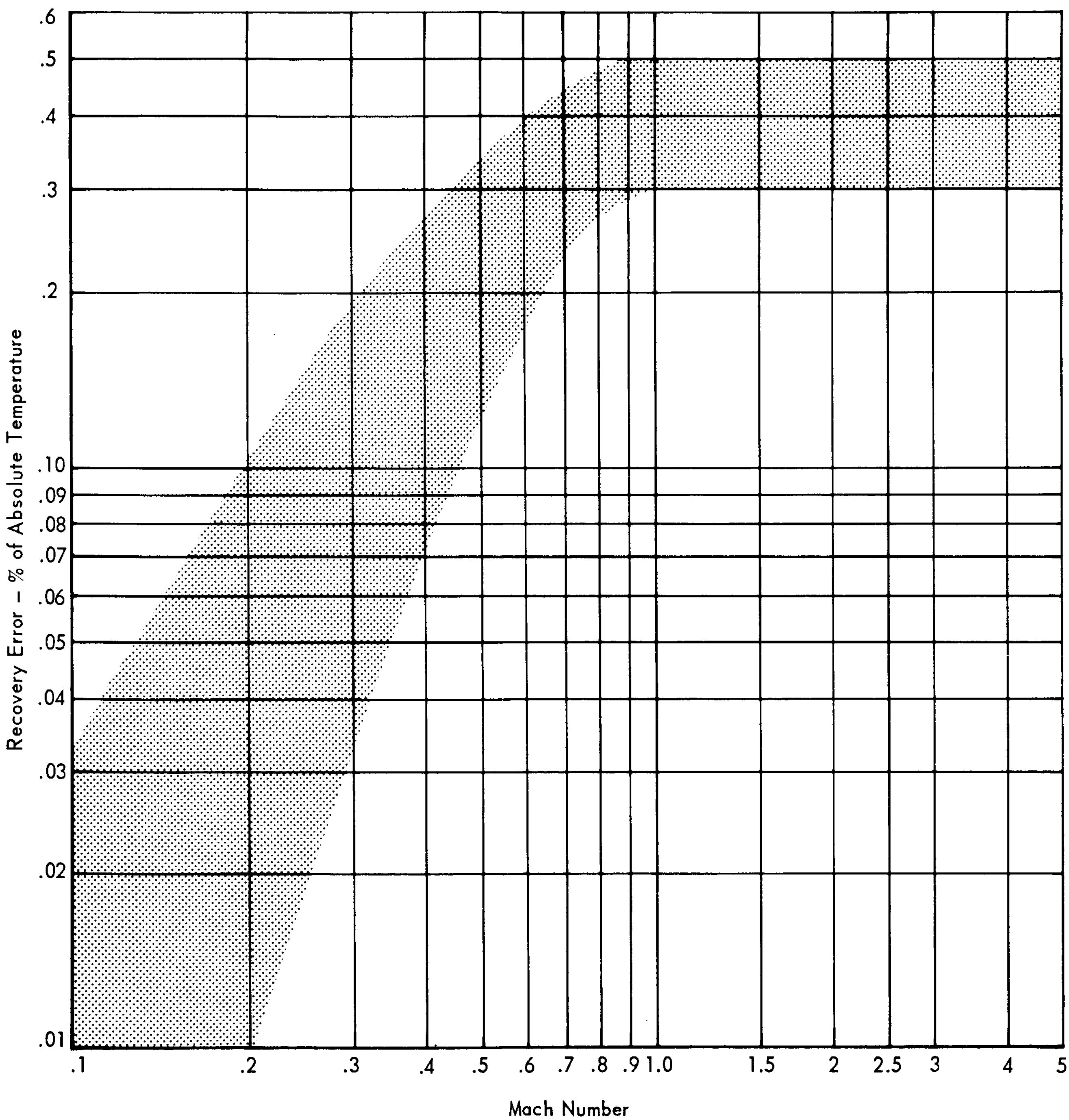


Figure 1.4–6

Both transducers use 0.01 watts of the 5.00 ±.05 Vdc power from the power subsystem and provide 0 to 40 mVdc outputs to the telemetry subsystem.

Figure 1.4-7 presents the stagnation transducer interface diagram and Figure 1.4-8 presents the base transducer interface diagram.

1.4.7 Reliability and Safety Considerations - The stagnation temperature transducer has a flow-blocking valve installed in the vent tube to keep temperatures at the sensing element to tolerable levels during peak heating portions of the entry thus increasing the instrument reliability.

There are no special safety considerations for these instruments.

1.4.8 Test Requirements - After the LV/PV mate an end-to-end check will verify temperature transducer integrity with the overall system.

No in-flight monitoring or checkout is required.

1.4.9 Development Requirements - All techniques and materials required to implement this instrument are existing. However, development tests are required:

- o to quantize angle-of-attack excursion effects
- o to quantize recovery errors as a function of Mach number
- o to determine the effects of terminal propulsion engine plumes on the base region transducer readings.

1.5 MASS SPECTROMETER

1.5.1 Equipment Identification and Usage - A quadrupole mass spectrometer is described here as representative of the instrument that will be used to measure the composition of the Mars atmosphere during descent and immediately after landing.

1.5.2 Design Requirements and Constraints - The design of this instrument is influenced by two primary factors; the anticipated atmospheric gas constituents to be measured and the gas inlet conditions. The mass range, sensitivity and resolution of the instrument determines the compounds it can detect in the atmosphere. The present mass range requirement is 10 to 60. The instrument resolution and sensitivity requirements are determined by the relative concentration of the compounds to be detected. The present sensitivity or detector range (and amplifier) requirement is assumed to be a 10^4 variation in detector current, while the resolution requirement is assumed to be 1 mass unit resolution for adjacent peaks with relative concentrations of 300 to 1. The variation in inlet conditions are approximated by the stagnation pressure and temperature time histories between Mach 5.0 and touchdown. Figures 1.5-1 and 1.5-2 contain stagnation pressure and temperature time histories for extremes of expected atmospheric models for the entry conditions shown. In addition to the above considerations, the interfaces between the instru-

STAGNATION TOTAL TEMPERATURE TRANSDUCER INTERFACE DIAGRAM

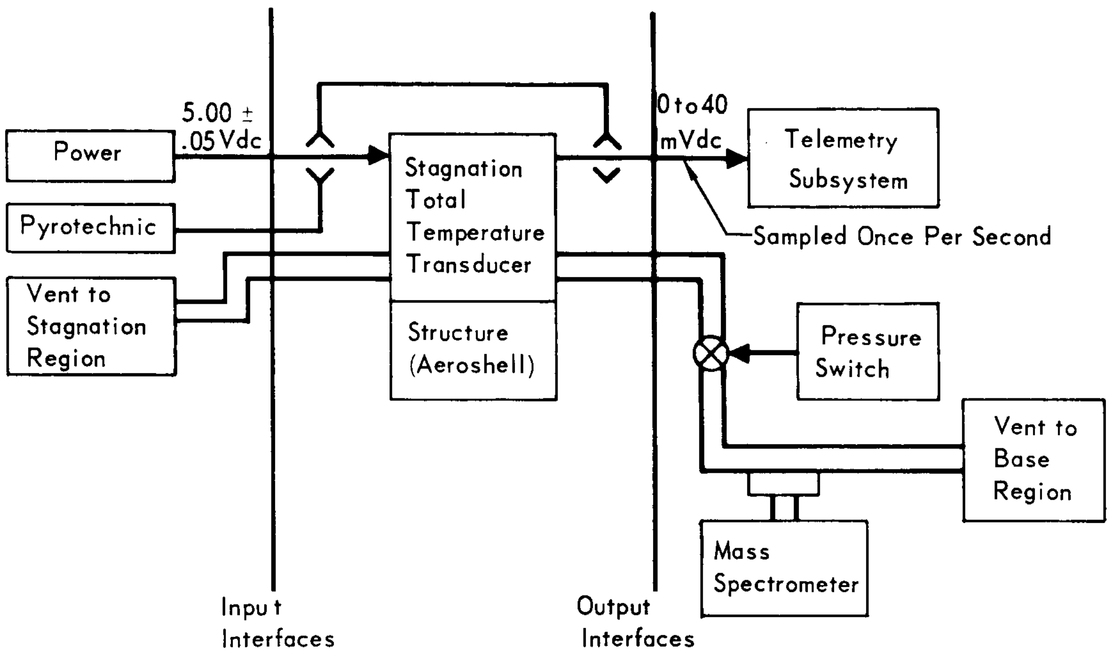


Figure 1.4-7

BASE REGION TOTAL TEMPERATURE TRANSDUCER INTERFACE DIAGRAM

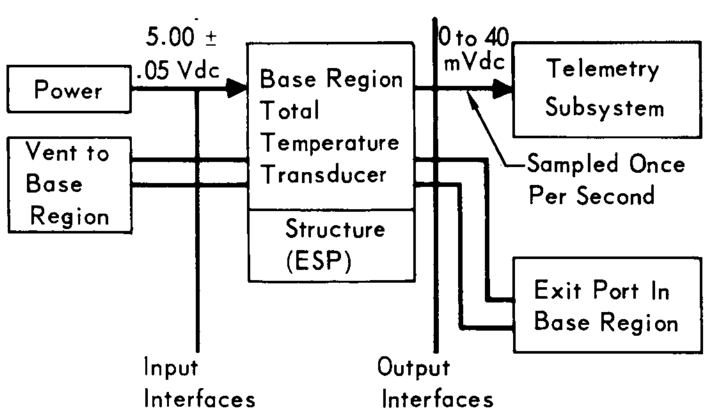


Figure 1.4-8

CALCULATED STAGNATION PRESSURE TIME HISTORY

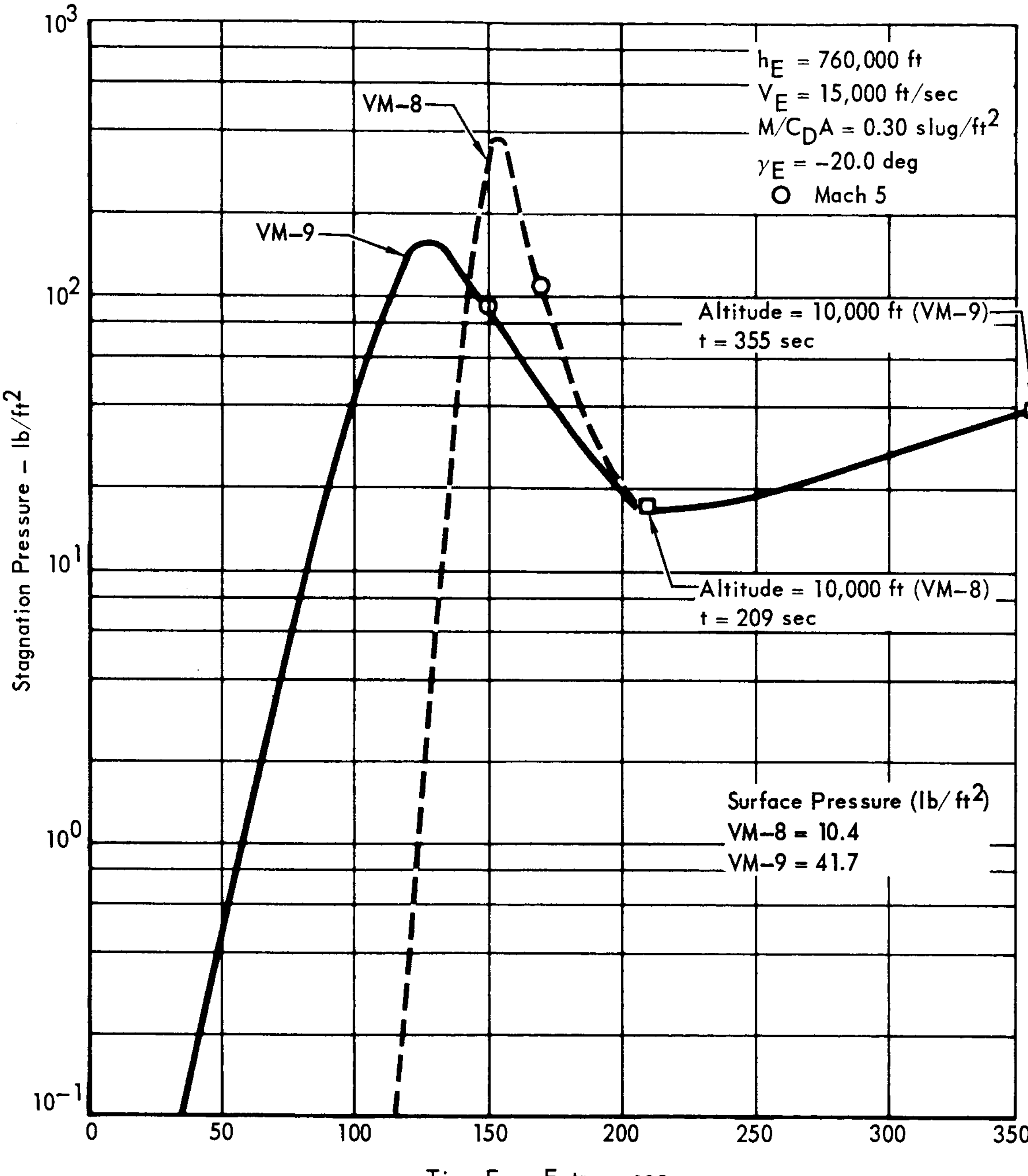


Figure 1.5-1

STAGNATION TEMPERATURE

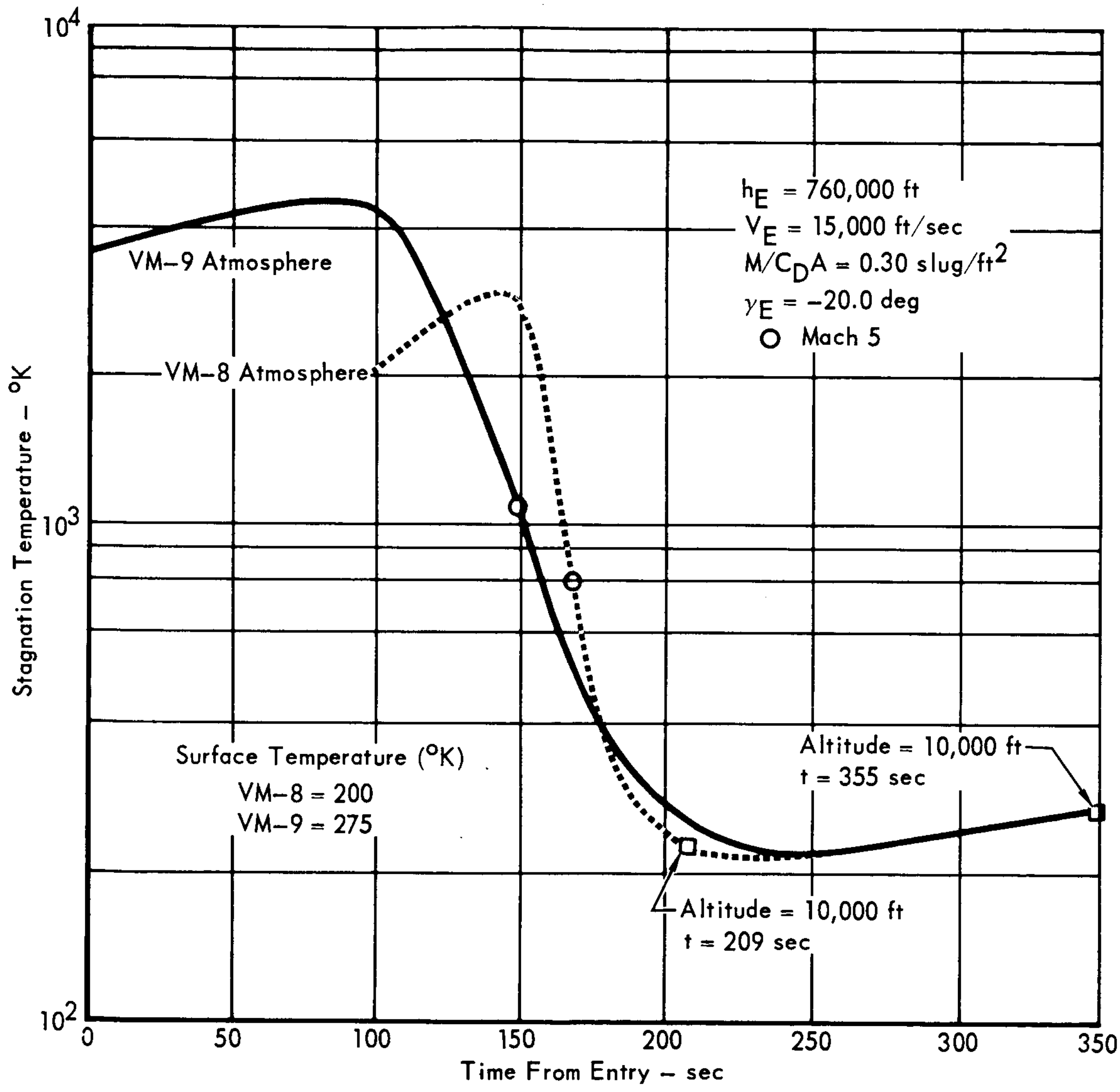


Figure 1.5–2

ment and the ESP subsystems must provide for mounting of the instrument, the supplying of electrical power and commands, the acceptance of output data for subsequent transmission, etc. The mass spectrometer, through its interface with the stagnation temperature outlet tube, imposes additional requirements on the design of the stagnation point gas entry port to insure that ablation product gases are prevented from entering the tube. Also, the technique used for separation of the stagnation pressure/temperature assembly from the tube must be one that insures continuance of flow through the tube after Aeroshell separation. To operate the instrument for pre-flight system integrity tests the instrument must be sealed and evacuated.

1.5.3 Physical Characteristics - Quadrupole mass spectrometer configurations have been developed and are presently undergoing development for space flight. The quadrupole instrument can be designed to have approximately the following physical characteristics:

Weight	8 lbs.
Size	2 x 7 x 14 inches
Volume	196 cu. in.

Various models presently available have components selected for high reliability and ability to withstand sterilization temperatures.

1.5.4 Operation Description - A simplified block diagram for the mass spectrometer is contained in Figure 1.5-3. At Mach 5.0 the inlet seal on the instrument is broken along with opening the valve that closes the stagnation temperature outlet tube. Atmospheric gas is brought to the mass spectrometer inlet by the stagnation temperature outlet tube (samples obtained from separate small capillary tube after touchdown). A continuous stream of gas molecules is brought into the instrument ion chamber from the outlet tube through the molecular leak. The atmospheric gas molecules are then ionized by electron bombardment from a hot filament electron source. The ionized molecules are then accelerated by accelerating electrodes and focused into a beam which is projected into the quadrupole filter. The electric fields in the filter are controlled such that only ions of specific charge to mass ratio are able to reach the detector, where they are neutralized by accepting electrons from the detector. The balance of the ions (those having charge to mass ratios resulting in unstable trajectories) are neutralized by collision with the quadrupole electrodes. The electric fields are varied such that ions of various

MASS SPECTROMETER BLOCK DIAGRAM

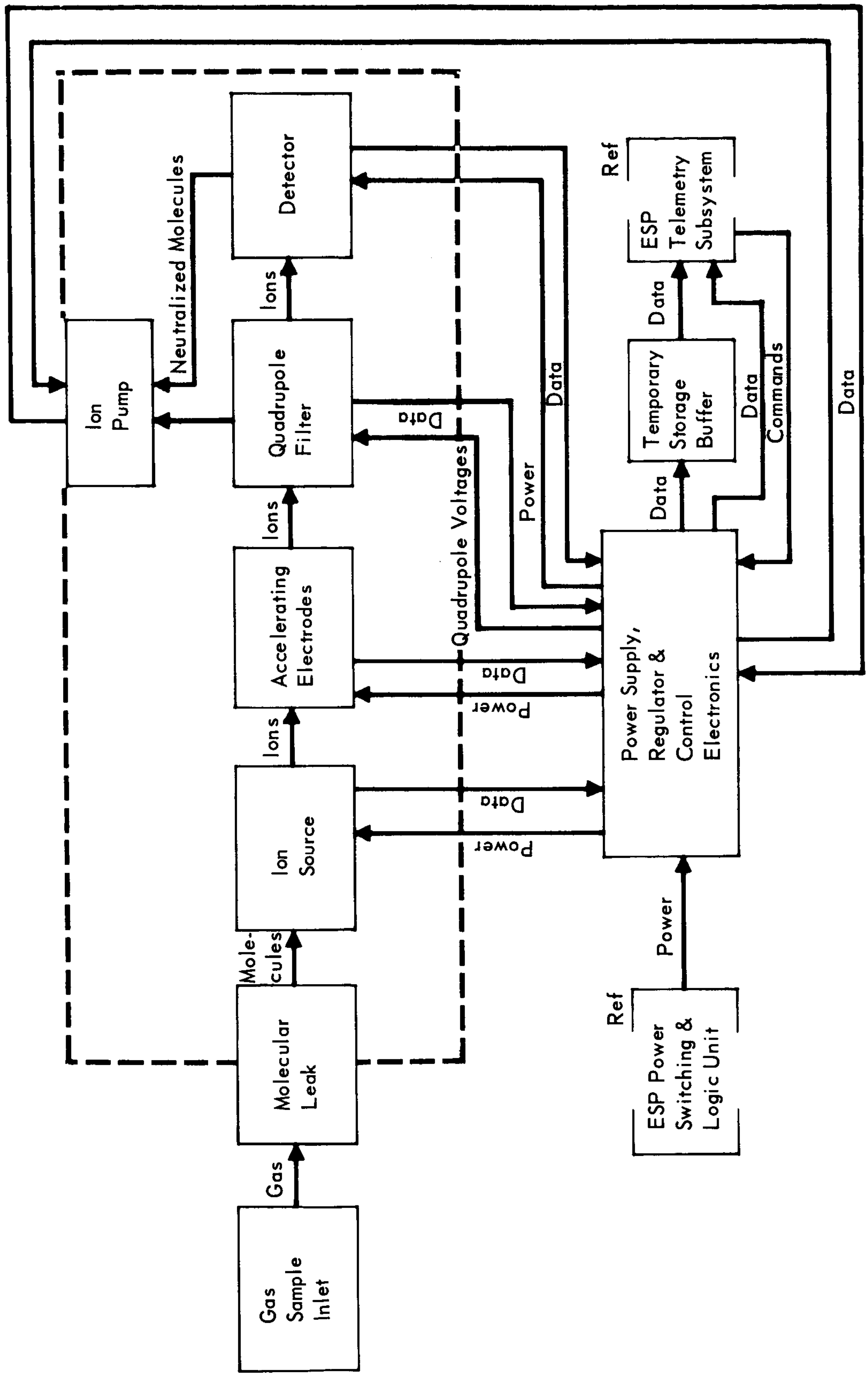


Figure 1.5-3

charge to mass ratios can pass through the filter and reach the detector at specific times. In this manner the entire mass range of the instrument is scanned (scan time is 2 seconds) and the amount of each particular charge to mass ratio ion constituent present is proportional to the detector current at a particular time in the scan. The detector current is sampled at the appropriate time within the scan and the current reading, corresponding to the concentration of ions of particular charge to mass ratio is converted to a digital word which is placed into temporary storage. After all readings of a particular scan are completed the data contained in temporary storage is read out to the data subsystem (which requires 8 seconds) after which another scan is begun, repeating the total above process on a 10 second cycle. An ion pump is used to remove the neutralized molecules and thus maintains the instrument internal operating pressure within the appropriate range.

1.5.5 Performance Objectives - With an instrument sensitivity or detector range of 10^4, atmospheric constituents present in concentrations as low as 100 ppm or 0.01 volume percent can be detected. Since the possible major constituents of the Mars atmosphere are presently considered to be nitrogen, argon and carbon dioxide, little difficulty with adjacent peak resolution is anticipated in determining major constituent composition. With conversion of the detector beam current analog signal to an 8 bit word (7 bit number with a 1 bit range) the accuracy is limited by the conversion to 0.785%. Thus, the molecular ion ratio or the ratio of partial pressure of a particular constituent to the total pressure accuracy cannot be better than 1.57%. The ranges and representative values for Mars atmospheric constituents taken from References (1) and (2) are presented in Figure 1.5-4.

If the oxygen concentration is in the range as shown, the instrument will be able to obtain a good measurement for oxygen. If not obscured by adjacent peaks, it also may detect the presence of water and some of the other trace elements, although the estimated concentrations shown are below the threshold of instrument sensitivity (0.01 volume %). Careful attention to materials selection and system cleanup or outgassing operations will be required to insure that any water detected is not Earth water that had been adsorbed on surfaces and walls of the system. By starting the mass spectrometer sampling at Mach 5, the problems associated with recognition of the primary atmosphere constituents after they have gone through disassociation and recombination processes are practically eliminated. For the worst entry conditions and atmospheric models, stagnation temperatures around $1000^{\circ}K$ may occur at Mach 5.0. At these temperatures, mixtures of carbon dioxide

CONSTITUENTS OF MARTIAN ATMOSPHERE

MAJOR CONSTITUENT (1)	MOL WT	COMPOSITION (% BY VOLUME) LOWER LIMIT	REPRESENTATIVE VALUE	UPPER LIMIT
N_2	28	0.0		80.
A	40	0.0		38.4
CO_2	44	9.5		100.
TRACE CONSTITUENTS (2)	**MOL WT**	**COMPOSITION (% BY VOLUME) LOWER LIMIT**	**REPRESENTATIVE VALUE**	**UPPER LIMIT**
O_2	32	0.01	0.05	0.15
N_2O	18	Trace	0.001	
O_3	48			
N_2O	44			
CH_4	16			
C_2H_6	30			
NH_3	17			

Note:

(1) From JPL model atmospheres
(2) From NASA CR 81307, 10 September 1965

Figure 1.5-4

and nitrogen may produce detectable amounts of nitric oxide. However, within 10 seconds or so later, the temperature will drop to where all reaction products will be below instrument detection thresholds.

1.5.6 Interface Definition - A structural mounting interface exists between the instrument and the ESP. A connection to the stagnation temperature outlet tube is required for instrument sampling. Thermal insulation will be required to reduce instrument heat losses and to isolate the instrument from other equipment. An interface diagram for the mass spectrometer is presented in Figure 1.5-5.

1.5.7 Reliability and Safety Considerations - The experiment reliability will be enhanced by use of multiple filaments in the instrument ionization chamber and keeping the system sealed and in an evacuated condition until sampling is begun.

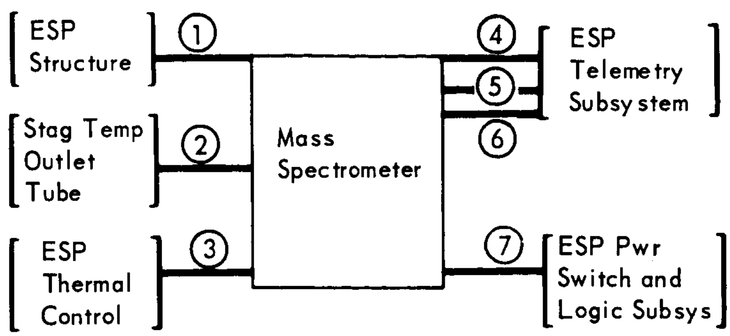
1.5.8 Test Requirements - Instrument validation tests will occur during pre-flight testing and late in the cruise phase or during pre-separation checkout. The instrument will be brought to temperature and pumped down using its internal ion pump. Also, at this time circuit integrity and voltage checks will be performed. At the end of the warm up and pump down period the instrument will be operated to determine internal pressures and composition of any residual gasses remaining in the instrument above its sensitivity threshold.

1.5.9 Development Requirements - Quadrupole mass spectrometers are presently in advanced stages of development which consists primarily of packaging and qualifying for flight requirements.

Reference (1) Final Report, Study of the Automated Biological Laboratory Project Definition, Volume III, NASA CR 81307 10 September 1965

(2) 1973 VOYAGER Capsule Systems Constraints and Requirements Document PD606-4, 12 June 1967

MASS SPECTROMETER INTERFACE DIAGRAM



INTERFACE	DESCRIPTION
1	Structural Mounting
2	Mechanical Attachment to Stagnation Temperature Outlet Tube
3	Thermal Interface
4	Commands: • Mass Scan Stop Control (Checkout Only) • Continuous Scan Controls (Checkout Only) • Scan Start • Data Readout • Break Inlet Seal • Clock • Switch to Capillary Inlet (at Touchdown)
5	Mass Spectrum Data Output (50–8 bit words per 2 second scan, readout in 8 sec, no readout during scan, 10 sec total cycle.)
6	Engineering Data Output: • B^+ Voltage • RF Voltage • Diode Temperature • Pump Pressure • Ion Chamber Temperature • Filament Number (Engineering data sampled 1/10 sec, synced to mass scanning)
7	Electrical Power Input (7 watts, 28.5 ± 5 Vdc)

Figure 1.5-5

SECTION 2

INSTRUMENTATION EQUIPMENT

This section presents the functional description of the instrumentation equipment required for acquisition of the ESP engineering performance and diagnostic data. The instrumentation equipment is identified as the portion of the telemetry subsystem where the data signals are sensed and conditioned to outputs compatible with the PCM encoders. The equipment configuration described herein is typical and evolved from a study of the ESP engineering measurement requirements. All components selected are potentially within the state-of-the-art and are based on designs which have been successfully used in previous spacecraft applications.

2.1 EQUIPMENT IDENTIFICATION AND USAGE - The instrumentation equipment consists of temperature sensors, signals conditioners, and associated power supplies.

2.1.1 Temperature Sensors - Temperature sensors are located throughout the ESP to provide temperature data from the ESP subsystems.

A preliminary list of ESP temperature sensors with descriptive functional and performance information appears in Figure 2-1. The list includes sensor part numbers used on other aerospace applications which are representative of those which will be used for the ESP. A total of 21 temperature sensors are required.

2.1.2 Signal Conditioners - The signal conditioners process analog, bi-level, and digital data signals from the various subsystems and supply inputs compatible with the PCM system 0 to 5 volt range. Signal conditioning of data signals is accomplished either in the originating subsystem equipment, in the telemetry equipment, or in a self-contained Signal Process Unit (SPU). Figure 2-2 contains an overall list of signal conditioning requirements including type and location. The signal conditioners located in the SPU are discussed in this section. The binary signal conditioning is located in the telemetry equipment and is, therefore, specified as part of that equipment. In addition, the subsystems will have built-in signal conditioning where the conditioning is most feasibly accomplished at the source. However, the same design philosophy of circuit isolation, impedance transformation and scaling applies to this category. The SPU contains a power supply which provides regulated dc voltages to the individual modules and also provides 5 V dc temperature sensor excitation.

2.2 DESIGN REQUIREMENTS AND CONSTRAINTS - the ESP instrumentation equipment for the most part will be modifications or evolutions of existing designs as influenced by the VOYAGER-peculiar constraints. Those principal constraints influencing

ESP INSTRUMENTATION – TRANSDUCER LIST

DATA IDENT. NO.	MEASUREMENT	SUBSYSTEM	RANGE	REPRESENTATIVE TRANSDUCERS						
				SIMILAR TO – MANUF/PART NO.	USED ON	TYPE	INPUT POWER	OUTPUT SIGNAL	SIZE	WEIGHT lb
Temperature										
RE 3–4	VHF Crystal Temperature (2)	Radio	25° to 150°F	RDF/55P886010–7	122Y	Thermo-resistive	5 Vdc, .4 mA	0–40m Vdc	1 x .8 x .4 Δ	.18
DE 1	SDS Oscillator Temperature	Telemetry	25° to 150°F	RDF/55P886010–7	122Y	Thermo-resistive	5 Vdc, .4 mA	0–40m Vdc	1 x .8 x .4 Δ	.18
DE 28	Data Storage Temperature	Telemetry	25° to 150°F	RDF/55P886010–7	122Y	Thermo-resistive	5 Vdc, .4 mA	0–40m Vdc	1 x .8 x .4 Δ	.18
TE 1–8	Equipment & Structural Temperature (8)	Thermal Control	–150° to 500°F	RDF/55P886010–5	122Y	Thermo-resistive	5 Vdc, .4 mA	0–40m Vdc	1.1 x .5 x .4	.123
EE 2	Battery Temperature	Power	0° to 120°F	RDF/55P886010–7	122Y	Thermo-resistive	5 Vdc, .4 mA	0–40m Vdc	1 x .8 x .4 Δ	.18
PE 10	Pressure Transducer Temperature	Science	–50° to 200°F	RDF/55P886010–5	122Y	Thermo-resistive	5 Vdc, .4 mA	0–40m Vdc	1.1 x .5 x .4	.123
VE 8–12	Entry TV Temperature (5)	Science	–50° to 200°F	RDF/55P886010–5	122Y	Thermo-resistive	5 Vdc, .4 mA	0–40m Vdc	1.1 x .5 x .4	.123
PE 13	Mass Spectrometer Diode Temperature	Science	–50° to 200°F	RDF/55P886010–7	122Y	Thermo-resistive	5 Vdc, .4 mA	0–40m Vdc	1 x .8 x .4 Δ	.18
PE 15	Mass Spectrometer Ion Chamber Temperature	Science	–50° to 200°F	RDF/55P886010–7	122Y	Thermo-resistive	5 Vdc, .4 mA	0–40m Vdc	1 x .8 x .4 Δ	.18

Note: Δ – Separate element and bridge

Figure 2–1

ESP INSTRUMENTATION – SIGNAL CONDITIONING LIST

DATA IDENT. NO.	MEASUREMENT	SUBSYSTEM	RANGE	SIGNAL CONDITIONING		
				LOCATION	TYPE	OUTPUT
RE 1–2	VHF Xmtr Power (2)	Radio		Radio Sub		0 to 5Vdc
RE 9–10	VHF Oscillator Drive (2)	Radio		Radio Sub		0 to 5Vdc
RE 11	Antenna Reverse Power	Radio		Radio Sub		0 to 5Vdc
RE 5–6	Xmtr Frequency (2)	Radio		Radio Sub		0 to 5Vdc
RE 7–8	Modulator Verification (2)	Radio		Radio Sub		0 to 5Vdc
DE 4	Telemetry Input Voltage	Telemetry	24 to 32Vdc	SPU	dc Signal Conv	0 to 5Vdc
DE 6–8	A/D Linearity Voltage (3)	Telemetry	0 to 5Vdc	–	–	0 to 5Vdc
DE 13	Cruise Commutator Input Voltage	Telemetry	24 to 32 Vdc	SPU	dc Signal Conv	0 to 5Vdc
DE 14–16	Cruise Comm ADC Linearity Volt (3)	Telemetry	0 to 5Vdc	–	–	0 to 5Vdc
DE 21–23	A-D Linearity Voltage (3)	Telemetry	0 to 5Vdc	–	–	0 to 5Vdc
DE 9–11	Low Level Amp. Linearity Volt (3)	Telemetry	0 to 40mVdc	–	–	0 to 40mVdc
DE 12	Transducer Voltage Source	Telemetry	0 to 40mVdc	–	–	0 to 40mVdc
DE 17–19	Cruise Comm Low Lev Amp Lin Volt (3)	Telemetry	0 to 40mVdc	–	–	0 to 40mVdc
DE 24–26	Low Level Amplifier Linearity Voltage (3)	Telemetry	0 to 40mVdc	–	–	0 to 40mVdc
DE 5	Slaved to Ref. Frequency	Telemetry	Bilevel	PCM	Binary	Bilevel
DE 20	Cruise Comm T/M Mode	Telemetry	4 bits	PCM	Binary	Digital
DE 2	Telemetry Mode	Telemetry	4 bits	PCM	Binary	Digital
DE 3	Vehicle Time	Telemetry	10 bits	PCM	Binary	Digital
DE 27	Telemetry Memory Readout	Telemetry		PCM	Binary	Digital
EE 1	Battery Voltage	Power	15 to 38Vdc	SPU	dc Signal Conv	0 to 5Vdc
EE 3	Battery Charge Current	Power	0 to 2A	SPU	Mag. Amp	0 to 5Vdc
EE 4	Battery Current	Power	0 to 6A	–	Binary	0 to 40mVdc
EE 5–6	Battery Switch Operation (2)	Power	Bilevel	PCM	Binary	Bilevel
VE 13–14	TV Target Voltage (2)	Science		Science		0 to 5Vdc
VE 17	TV Light Level	Science		Science		0 to 5Vdc
VE 15–16	TV Erase Lamp Status (2)	Science	Bilevel	Science	Binary	Bilevel
PE 11	Mass Spect. – B+	Science		Science		0 to 5Vdc
PE 12	Mass Spect. – RF Voltage	Science		Science		0 to 5Vdc
PE 15	Mass Spect. – Filament Voltage	Science		Science		0 to 5Vdc
PE 17	Mass Spect. – Detector Range	Science		Science		0 to 5Vdc

Figure 2–2

the modifications are sterilization, low power consumption and satisfactory operation after a dormancy period of approximately 6,000 hours. Specific requirements for the three categories of instrumentation equipment are delineated below:

2.2.1 Temperature Sensors - Structure, equipment, and component temperature measurements are required to provide temperature time histories during all mission phases of ESP operation for:

a. Verification of proper thermal control performance
b. Verification that temperature-critical electronic components and batteries are maintained within the controlled temperature spans.
c. Indication of critical electronic component operating temperatures
d. Diagnosis of failures and failure mode determination

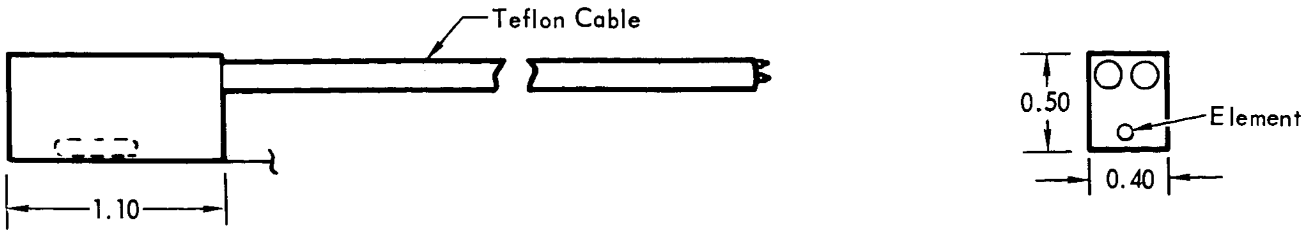
2.2.2 Signal Process Unit - Signal conditioning is required for monitoring the power system bus voltage, battery charge current, and telemetry input voltages during all mission phases. The requirement of the instrumentation power supply is to convert unregulated 28 Vdc battery power to a precision 5 Vdc with high efficiency and long term stability during all mission phases.

2.3 PHYSICAL CHARACTERISTICS -

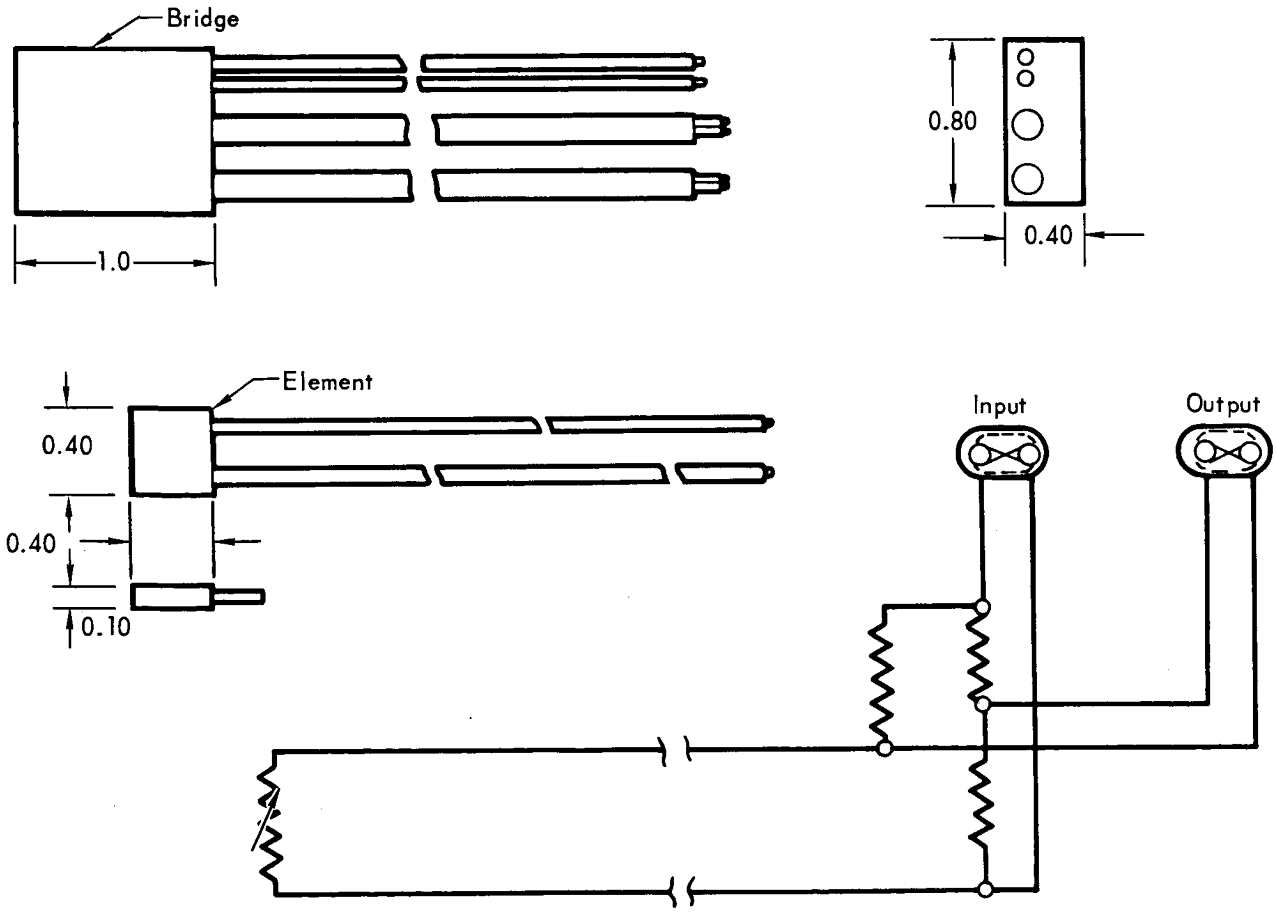
2.3.1 Temperature Sensors - The temperature sensors will be of two types, integral bridge and element or separate element and bridge, connected in a conventional wheatstone bridge arrangement. The integral bridge sensors have a platinum resistance element and bridge completion network molded into one integral unit. The sensing elements are fully annealed pure platinum wire mounted in a strain free manner. Where it is not possible to mount an integral bridge sensor a separate element and remote bridge will be used. Depending on their locations, surface temperature sensors are attached by cementing or spot welding of flanges. Silver doped cement is used for good heat transfer. Outline drawings of typical temperature sensors are illustrated in Figure 2-3.

2.3.2 Signal Conditioning - The ESP Signal Process Unit (SPU) is a self-contained unit consisting of plug-in, solid state, signal converter modules and associated power converter modules; interconnecting wiring; interface connectors; necessary hardware; and a suitable enclosure. The configuration of the SPU is governed by the quantity of signal converter modules needed for ESP measurement requirements. A preliminary estimate of the SPU size is 6 modules in a 6" x 4" x 4" configuration weighing approximately 3 pounds. The SPU will be designed for flexibility to accommodate changes in quantity, type and range of measurements. Each signal converter module will have standardized dimensions and connector pin assignments for

REPRESENTATIVE TEMPERATURE SENSORS



55P886010-5 Temperature Sensor/Integral Bridge & Element



Wiring Diagram

55P886010-7 Temperature Sensor/Remote Element & Bridge

Figure 2-3

interchangeability. The self-contained power supply operates from the 28 Vdc source. The total SPU power requirement is estimated at 4 watts.

2.4 OPERATION DESCRIPTION - The block diagram of the instrumentation equipment is shown in Figure 2-4.

2.4.1 Temperature Sensors - Temperature sensors are provided to sense the equipment temperatures as detailed in Figure 2-1 and to convert these to proportional electrical outputs ranging from 0 to 40mVdc during the interplanetary cruise and continuing through all mission phases to Mars surface impact.

2.4.2 Signal Conditioners - DC signal converter modules and a magnetic amplifier module are provided in the SPU to convert electrical inputs from the ESP power and telemetry subsystems into signals compatible with the PCM encoders. The dc signal converters are voltage monitors designed to meet the range of parameters being monitored, and convert these voltages into proportional 0 to 5 Vdc outputs. The SPU will be in operation intermittently during interplanetary cruise when commanded by the telemetry equipment and in operation continuously beginning at pre-separation checkout until Mars surface impact.

ESP INSTRUMENTATION EQUIPMENT BLOCK DIAGRAM

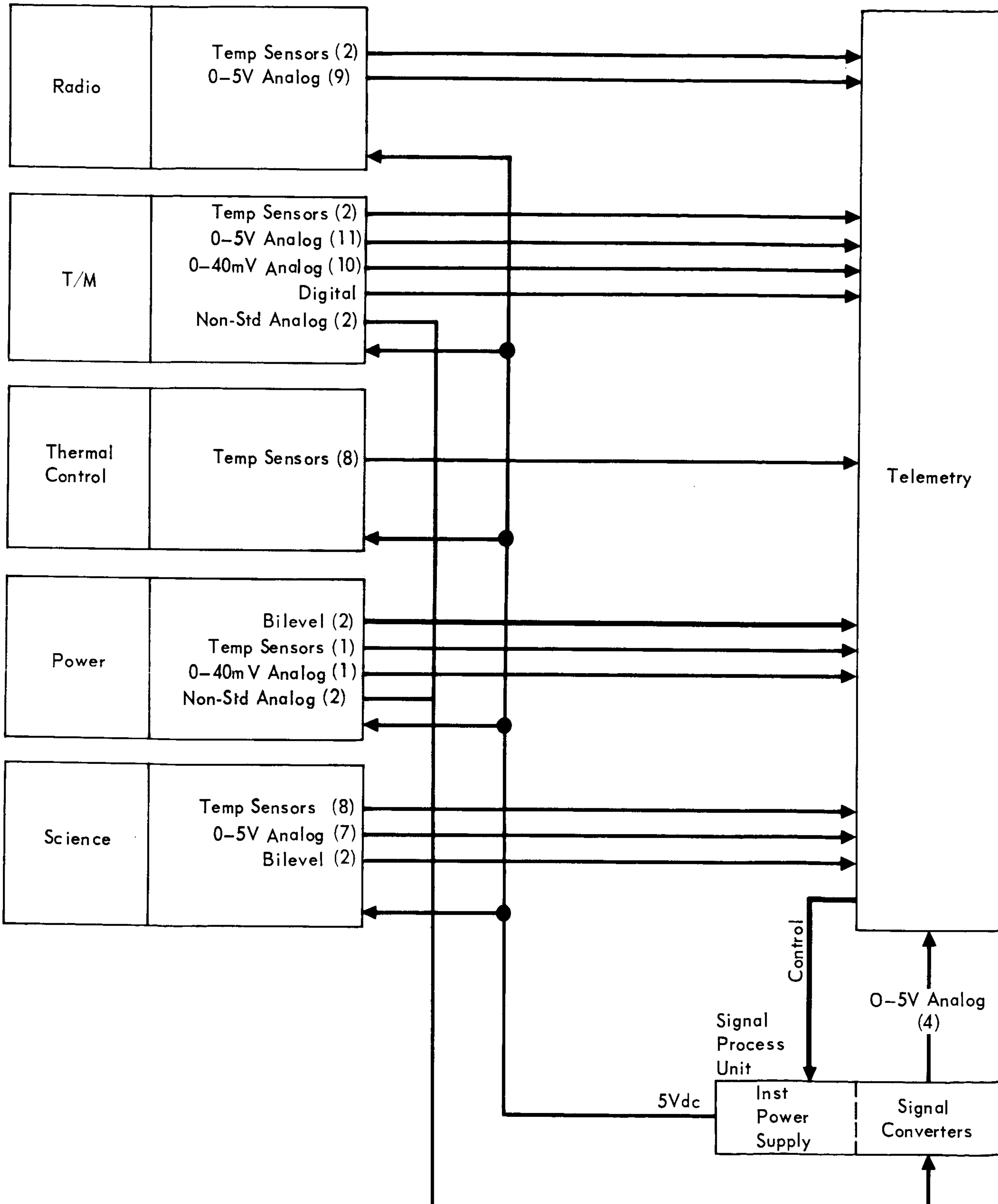


Figure 2-4

2.5 PERFORMANCE CHARACTERISTICS -

2.5.1 Temperature Sensors - The temperature sensor performance summary is given below in Figure 2-5.

FIGURE 2-5 TEMPERATURE SENSOR CHARACTERISTICS

- o Exitation - 5.0 ± .005 Vdc
- o Power Consumption - 2 mW Average
- o Output Signal - 0 to 40 mVdc
- o Theoretical Transfer Function - Linear
- o Output Impedance - 500 ohms max.
- o Static Error Band - 2% FS

Figure 2-5

2.5.2 Signal Conditioners - The Signal Process Unit (SPU) performance summary is tabulated in Figure 2-6.

FIGURE 2-6 SPU PERFORMANCE CHARACTERISTICS

- o Input Power - 30±6.5 Vdc
- o Total Power Consumption - 4 watts
- o dc Signal Converters
  - Input Signal Impedance - 500 kilohms
  - Output Signal - 0 to 5 Vdc
  - Output Impedance - 2 kilohms m
  - Error - 1% FS max.
- o Temperature Sensor Excitation Power Supply
  - Output Voltage - 5.0 Vdc
  - Output Current - 25 mA
  - Regulation - 0.20%
  - Ripple - 10 mV peak to peak

Figure 2-6

2.6 INTERFACE DEFINITION - Refer to Figure 2-4 for interface identification.

2.7 RELIABILITY CONSIDERATIONS - The estimated reliability factor for the ESP Instrumentation Equipment is 0.993 for mission success. A primary objective in the selection of the Instrumentation Equipment design concept was that no instrumentation failure would cause a significantly degrading or possible catastropic direct effect on any functional ESP subsystem. This objective will be achieved by providing a high conditioner to source impedance ratio which is present at all times, that is, isolation will still be provided in the event of a signal conditioner failure or shut down. The reliability goal will be achieved by using components with a history of high reliability, by operating the components at minimum electrical stress, and by providing heat sinks for low operating thermal stress. Sturdy mounting of ruggedized sensors will be required to assure long duration operation.

While instrumentation failures must be minimized the system should be capable of tolerating some individual instrument failures. For failure tolerance, two techniques have been considered: backup redundant sensor and direct data correlation between existing sensors. An example of the latter technique is temperature monitoring. ESP testing will establish correlation between individual temperature measurements. Thermal testing is effective in correlation of temperature distribution throughout the ESP and therefore will provide indirect data retrieval from a failed temperature sensor during a mission.

2.8 TEST - All modules in the SPU utilizing active elements will provide circuitry for removing input signals and applying at least two known precision input voltages for pre-flight and in-flight calibration and operational confidence checks.

The temperature sensors utilize only passive circuit elements and will not be automatically calibrated during pre-flight and in-flight checks. These units are made up of fixed resistors of the highest quality available. The basic sensors will be more reliable than the automatic checking equipment required for remote calibration.

2.9 DEVELOPMENT REQUIREMENTS - Instrumentation items presently available are potentially capable of meeting the VOYAGER mission requirements. However, the state-of-the-art will be strained in several areas to meet sterilization and long term stability requirements. Such materials as potting, varnish, insulation, etc., and soldering and assembly techniques used in manufacturing present day transducers and signal conditioners will require changes to meet the stringent VOYAGER requirements. These items will require investigation and development testing.

SECTION 3

DATA STORAGE SUBSYSTEM

The Entry Science Package Data Storage Subsystem consists of the ESP Data Storage Assembly (Paragraph 3.1) and the Spacecraft-Mounted ESP Support Data Storage Assembly (Paragraph 3.2). The ESP DSA provides time-delay functions, and the FSC-MTD ESP Support DSA provides tape storage of relayed ESP data, which arrive at the FSC at a rate too high to be relayed to Earth in real time.

3.1 ENTRY SCIENCE PACKAGE DATA STORAGE ASSEMBLY

3.1.1 Equipment Identification and Usage - The ESP Data Storage Assembly accepts data from the real time data interleaver of the ESP TM equipment and provides two lines of delayed data to the delayed data interleavers of the TM equipment. Figure 3.1-1 is a functional block diagram of the ESP Data Storage Subsystem.

The ESP Data Storage has two coincident-current core memories. The first provides a nominal 50 seconds of delay, the output feeding both the data interleaver, and the second memory which provides an additional nominal 100 seconds of delay for a nominal total of 150 seconds of delay. The assembly consists of the following subassemblies:

- o 2048 word by 23 bit core memory
- o 4096 word by 23 bit core memory

Each core memory will consist of a 3D organized core stack and the electronics necessary to drive it. The data storage assembly is driven at a 910 bps rate, and the actual delays are 51.8 sec and 103.5 sec for a total delay of 155.3 sec.

3.1.2 Design Requirements and Constraints - The input data rate is 910 bps. The design requirements call for data to be delayed 50 seconds and 150 seconds. At least 45500 bits of storage are needed for 50 seconds of delay. Similarly the 100 second delay needs 91000 bits. The Data Storage Assembly must be compatible with the bit stream moving through the data interleaver. Therefore, the Assembly must operate in a serial by bit mode. Following is a summary of design requirements:

Total Storage Capacity	136500 bits
Read Rate	910 bps
Write Rate	910 bps

Other requirements and constraints on the design of the ESP Data Storage have been called out in Reference 3.1.2-1.

ESP DATA STORAGE FUNCTIONAL BLOCK

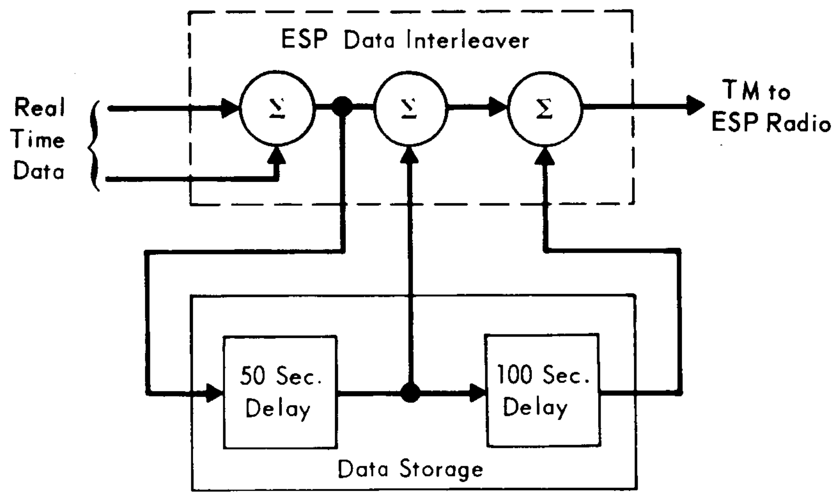


Figure 3.1-1

3.1.3 Physical Characteristics - The 50 second delay memory requires a volume of 120 in^3, weighs 3.5 lbs and consumes 3 watts of power. The 100 second delay memory requires a volume of 132 in^3, weighs 4 lbs, and consumes 4 watts. This gives subsystem totals of 252 in^3, 7.5 lbs, and 7 watts. The power consumed is an average based upon the use of a 20 microsecond cycle time at a bit rate of 910 bps.

3.1.4 Operation Description - The Data Storage Assembly is used during entry and terminal descent to provide storage and retransmission of all data except TV. The operation of both delay memories is identical. They are coincident-current units utilized in a serial by bit configuration. The memory cycle is the split cycle or read-modify-write cycle. Old data is read out of an address and new data is written into that address before the address register is advanced. The delay of each memory is proportional to the storage capacity. Figure 3.1-2 is a block diagram of the ESP Data Storage Assembly. Following is a description of the elements in the 50 second delay memory.

3.1.4.1 Internal Timing Logic - This logic provides all timing pulses necessary for the transfer of data in and out of the storage array. The ESP clock generator is used as the memory master clock. From this clock the timing pulses for the read, write, and inhibit current pulses are generated. The sense amplifier strobe and address counter advance are also derived from the clock. The internal timing logic is enabled only when the Mod 23 counter overflows. Figure 3.1-3 is a diagram of the memory internal timing.

3.1.4.2 MOD 23 Counter - The MOD 23 counter counts the number of bits shifted into the input register. The overflow of this counter enables the memory cycle.

3.1.4.3 Address Register - This register is a binary counter that advances at the end of each memory cycle. For the 50 second delay it is a 11 bit counter. A 12 bit counter is required for the 100 second delay.

3.1.4.4 X - Y and Inhibit Decode and Drivers - Each dimension of the core stack requires a decode matrix and current drivers that provide the half select currents to drive the core stack. The X - Y currents are present during both read and write. The inhibit current is only present when it is desired to write a zero in an address.

3.1.4.5 Storage Array - The Storage array for the 50 second delay will be a 3D organized stack using 20 mil. O.D. cores with 2048 cores per plane and 23 planes for a total of 47104 bits of storage capacity. The inhibit and sense lines are unique to each plane and the X - Y lines are common to all planes in the stack. The 100 second delay will require 4096 cores per plane, and 23 planes giving a total of 94208 bits of storage.

ESP DATA STORAGE BLOCK DIAGRAM

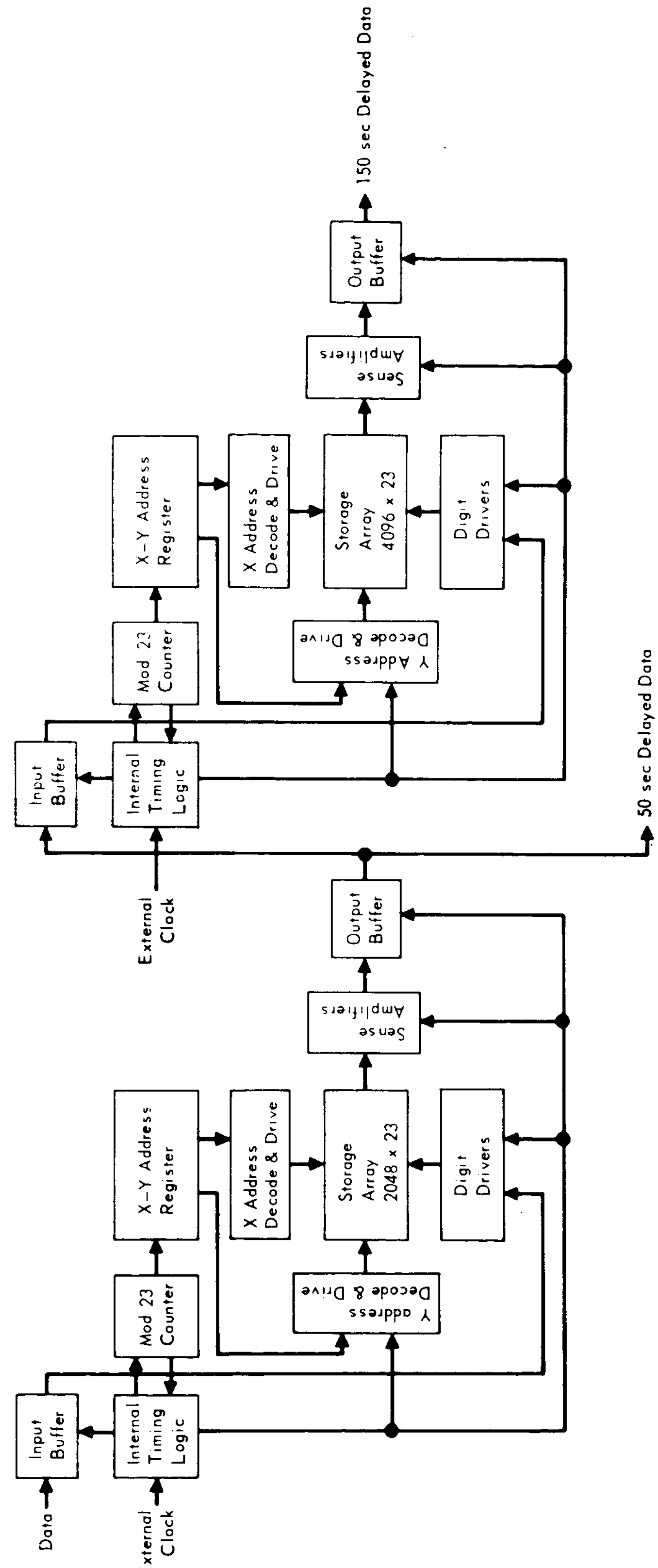


Figure 3.1-2

MEMORY INTERNAL TIMING

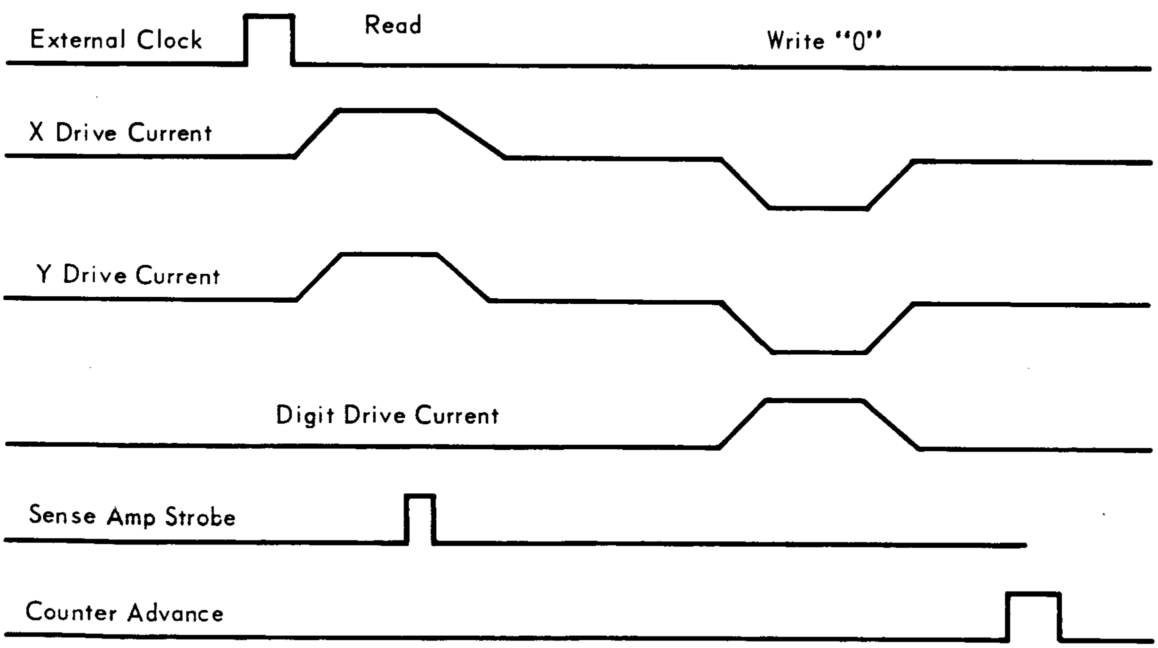


Figure 3.1-3

3.1.4.6 Sense Amplifiers - The output of a 20 mil. core is on the order of 30 mV. Therefore, high gain differential amplifiers are required to bring the data contained in the cores up to standard integrated circuit logic levels. One sense amplifier is required for each bit in a word. This gives a total of 23 sense amplifiers in each memory.

3.1.4.7 Output Buffer - A 23 bit parallel-in-serial-out shift register accepts data as it comes out of core in parallel and shifts the data out serially.

3.1.5 Performance Characteristics - ESP data storage performance characteristics are listed in Figure 3.1-4.

3.1.6 Interface Definition - The ESP Data Storage Assembly has signal interfaces with the ESP power subsystem and the ESP telemetry equipment as given on the Interface Diagram, Figure 3.1-5. The digital data streams move at 910 bits per second.

3.1.7 Reliability and Safety Considerations - The following subparagraphs describe the reliability and safety considerations of the ESP Data Storage Assembly.

3.1.7.1 Mission Success Definition - The data storage provides delayed storage for the interleaved CB and low-rate ESP data. There are two delay periods required: nominally 50 seconds and 150 seconds.

3.1.7.2 Reliability Model - ESP data storage has an assessed reliability factor of 0.992. Figure 3.1-6 shows the reliability block diagram.

3.1.7.3 Mission Failure Modes and Effects - Figure 3.1-7 shows the failure fault tree for the ESP data storage.

Redundancy is not employed within the data storage.

The degraded modes of operation of the Data Storage Subsystem can be categorized as follows:

a. Correct 50 sec delayed data and erroneous 150 sec delayed data.

b. Erroneous 50 sec and 150 sec delayed data.

This first mode would result from a failure in the 100 second delay storage unit. The second mode would result from a failure of the 50 second delay unit. The erroneous data can be either no data, incorrect data, or a data delay of a time period other than specified. Operation in either of the degraded modes will not cause mission failure because the Capsule Bus provides functionally redundant data transmission.

PERFORMANCE CHARACTERISTICS, ESP DATA STORAGE

CHARACTERISTIC	50 SECOND DELAY	100 SECOND DELAY
Storage Capacity	47,104 bits	94,208 bits
Read Access Time	0.7 μ sec	0.7 μ sec
Write Access Time	0.85 μ sec	0.85 μ sec
Split Cycle Time	2.0 μ sec	2.0 μ sec
Bit Error Probability	$< 10^{-4}$	$< 10^{-4}$
Power	3.0 watts	4 watts

Figure 3.1-4

ESP DATA STORAGE INTERFACE DIAGRAM

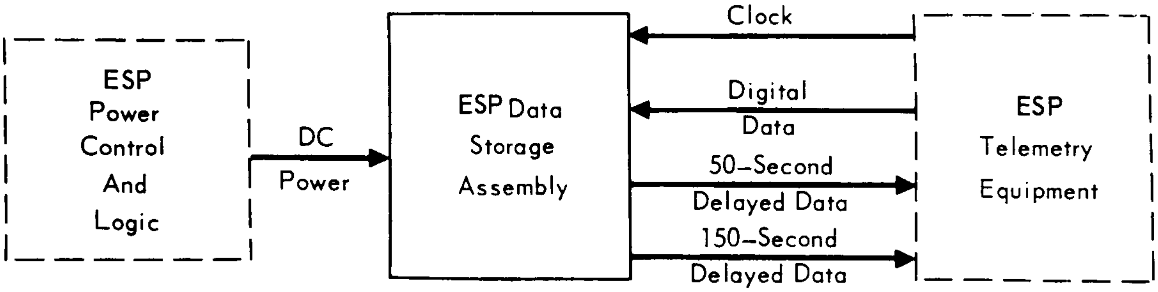
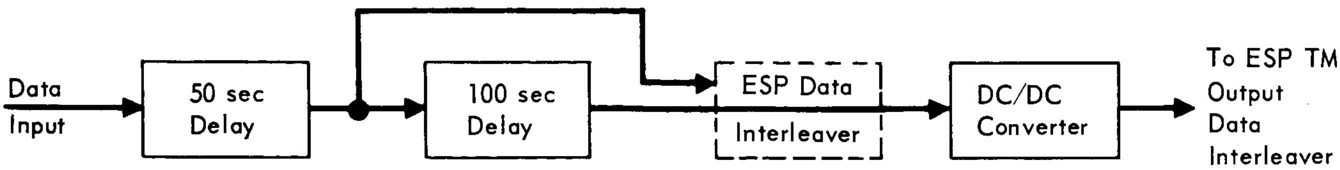


Figure 3.1-5

ENTRY SCIENCE PACKAGE
DATA STORAGE RELIABILITY MODEL



$$P_s \text{ (Delayed Data, 115 hr)} = e^{-0.008060} = .991972$$

Figure 3.1-6

ESP DATA STORAGE RELIABILITY FAULT TREE

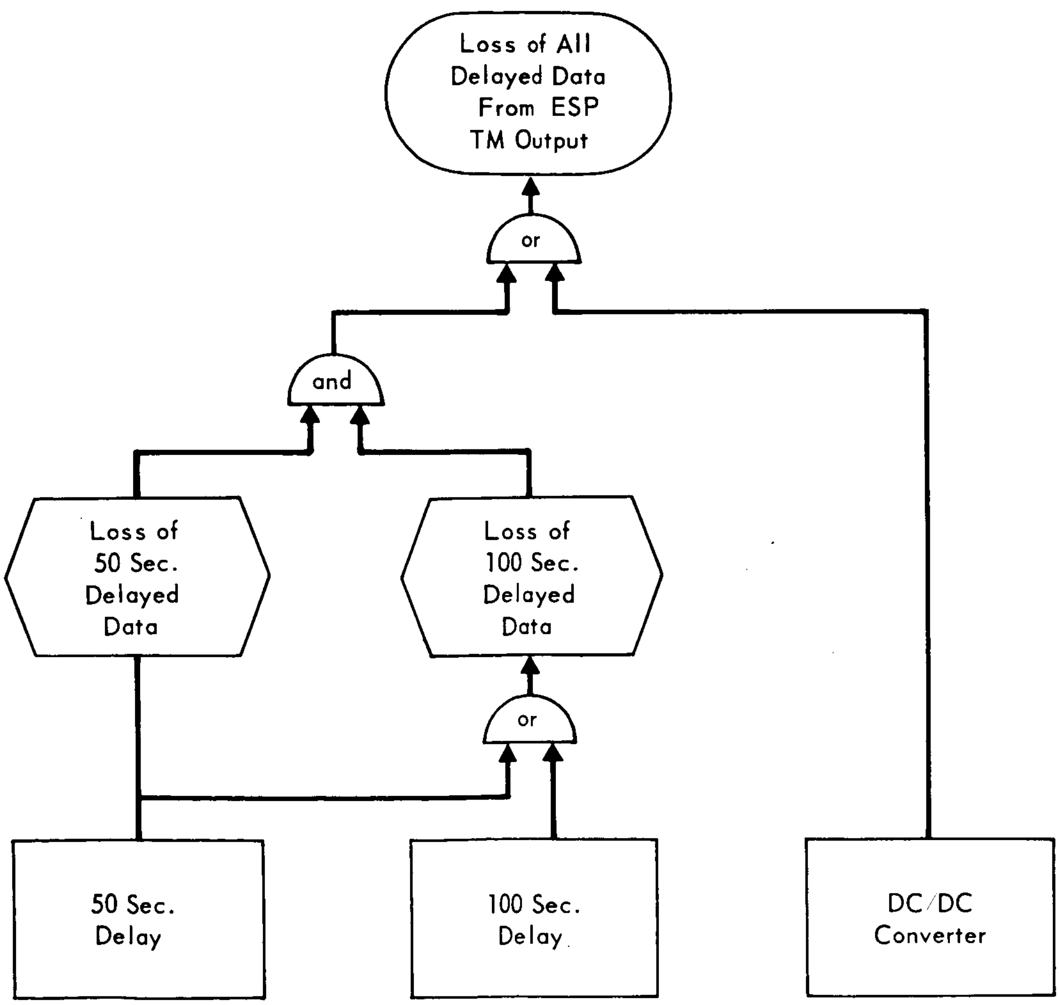


Figure 3.1-7

3.1.7.4 Complexity Estimate - The ESP Data Storage Assembly is estimated to have a total piece parts count of 149,775, of which 147,456 are magnetic cores. The estimated parts count is detailed in Figure 3.1-8.

3.1.7.5 Safety Considerations - No high voltages or pyrotechnics are employed.

3.1.8 Test - Tests (See Figure 3.1-9) shall be performed prior to canister enclosure, after enclosure prior to final sterilization, after sterilization, and during pre-launch operations. All ground testing shall be performed with the use of Operational Support Equipment (OSE). In-flight checkout (pre-separation) is initiated by the appropriate flight capsule test programmer upon command from the DSIF and monitored through the spacecraft telemetry system.

Ground test of the Data Storage Assembly is accomplished by inserting a data pattern, and verifying the pattern as it comes out of memory.

In-flight testing is accomplished as part of an end-to-end check of the telemetry subsystem.

3.1.9 Development Status - Core stacks and the ancillary peripheral electronics that have been used for spacecraft applications are available as off-the-shelf items. Consultation with the designers and manufacturers of these devices has indicated that these assemblies should be capable of withstanding the sterilization and decontamination environments without degradation. Qualification of parts and suppliers will require tests which stress these environments in addition to requirements for parametric stability, long life, and failure rate.

ESP DATA STORAGE ASSEMBLY
RELIABILITY PARTS COUNT ESTIMATE

PART TYPE	FAILURE RATE, BITS	COMPONENT PARTS COUNT			TOTAL PARTS	TOTAL FAILURE RATE, BITS
		DC/DC CONV.	50 sec DELAY	100 sec DELAY		
Integrated Ckts:						
Digital	10		42	60	102	1,020
Linear	30		24	24	48	1,440
Transistors:						
GP	5.0	2	120	176	298	1,490
Power	50	4			4	200
Diodes:						
GP	1.1	8	294	420	722	794.2
Zener	10	2			2	20
Resistors:						
Metal Film	0.3	10	202	288	500	150
Carbon Comp.	0.1		155	221	376	37.6
Capacitors:						
Ceramic	1.0	4	36	52	92	92
Solid Tant.	2.0	4	28	28	60	120
Cores	0.002		49,152	98,304	147,456	294.9
Connections	0.1		(3,150)[1]	(4,500)[1]	(7,650)[1]	765
Transformer, LV	5.0	1			1	5
Transformer, Sw	5.0		46	66	112	560
Inductors, LV	10	2			2	20
Totals		37	50,099	99,639	149,775	7,008.7

Notes: 1 – The number of core connections contributes to the failure rate only and is not included in the number of parts.

Figure 3.1-8

ESP DATA STORAGE
TEST MATRIX

TEST	Non-operative Test	CB Data Storage	Power Supply Voltage	Power Supply Current	Input Data	Memory Operation	Memory Control Signals	Output Data (50 sec Delay)	Output Data (150 sec Delay)
ACCY REQD			3%	3%	Digital	Digital	Digital	Digital	Digital
Telemetry Monitor			X	X					
System Test (Pre-canister)			X	X	X	X	X	X	X
System Test (With Canister)			X	X	X	X	X	X	X
Pre-launch			X	X	X	X	X	X	X
In-flight Checkout			X	X	X	X	X	X	X

Figure 3.1-9

REFERENCES

3.1.2-1 - <u>1973 VOYAGER Capsule Systems Constraints and Requirements Document.</u> California Institute of Technology, Jet Propulsion Laboratory, 1 January 1967 and May 1967.

3.2 Spacecraft Mounted Entry Science Package Support Data Storage -

3.2.1 Equipment Identification and Usage - The Spacecraft Mounted Entry Science Package (S/C-MTD ESP) Support Data Storage Assembly (DSA) is composed of a magnetic tape recorder/reproducer and interface equipment with a total storage capacity of $\approx 30 \times 10^6$ bits. The DSA records engineering and science data, including TV, which are transmitted from ESP to S/C via relay link during Flight Capsule entry and descent. These data are recorded on magnetic tape and are available for delayed playback and transmission at rates compatible with the Flight Spacecraft Telemetry and Radio Subsystems.

3.2.2 Design Requirements and Constraints - The S/C - MTD ESP Support DSA is constrained by the mission requirements to remain in a power-down condition for extended periods. This condition requires the use of a tape transport with an Iso-Elastic drive or equivalent transport design which inhibits tape spillage during a power-down period.

Total storage capacity of $\approx 30 \times 10^6$ bits of digital data is required with serial data rates of 55,860 bits per second. This storage capacity and data rates are best achieved with a multi-track recorder and serial to parallel conversion with parallel recording of data for an effective reduction in bit rates. This will minimize size, weight, and power requirements of the S/C MTD Support DSA through a reduction in the required tape capacity and operating speed.

Playback of stored data will be provided at a rate compatible with the FSC Telemetry subsystem. This is accomplished by phased-locked loop control of the playback drive and provides data to the telemetry system in synchronism with the FSC Telemetry clock. Output data is obtained from a temporary storage register (buffer) which also provides the reconstruction of the data train from a parallel to serial format.

These requirements are based on the equipment usage as defined in Paragraph 3.2.1. This recorder will be available to the Flight Spacecraft System following landing operations of the Flight Capsule; provided that additional requirements derived from that application do not compromise the primary mission of the recorder. Consequently, requirements can be expected to change and will be resolved in Phase C.

3.2.3 Physical Characteristics - The DSA occupies 350 cubic inches, weighs 10 lbs., and requires 10 watts, when operating. It records data on 14 tracks and contains 200 feet of 1-mil heavy duty Mylar-backed instrumentation tape. The tape transport is continuous motion reel-to-reel, using flangeless reels and a peripheral belt-driven configuration.

3.2.4 Operation Description - The tape transport configuration includes a single seamless Mylar drive belt of approximately the same width as the tape which encircles the periphery of the tape packs and is driven by a differential capstan system minimizing tape dropouts due to tape/head separations. The tape pack remains under positive control of the pressure belt to prevent tape spillage from the loss of tension in the standby mode with power off.

Data are recorded in one complete unidirectional tape pass, and reproduced with the tape driven in the opposite direction.

The DSA block diagram is shown in Figure 3.2-1. Serial data at 55,860 bps are loaded into a 14-bit memory, and the memory contents clocked on to the tape. The DSA is commanded into the record mode by control signals which originate within the S/C-MTD ESP Radio bit synchronizer. The DSA records only when data is being received from the ESP during Entry and Terminal Descent, under automatic control of the link rather than the FSC CC&S. A single, recording speed of 3.99 inches per second provides the record capability at the adjusted bit rate with parallel recording on 14 tracks.

Playback is under control of the FSC - CC&S. It is probable that the FSC will transmit data at 7200 bps, requiring a record/playback ratio of 7.76 to 1.

During playback, a phase-locked loop motor drive provides precise control of tape speed resulting in output data in synchronism with telemetry. The playback motor speed is controlled by phase comparison of the reproduced clock signal with the FSC telemetry clock. Data is shifted out of the 1-stage output register in exact synchronism with the FSC clock eliminating the time displacement errors associated with digital recording.

Backup control of recorder playback is provided by the FSC Command Subsystem.

The S/C MTD ESP Support DSA will include as part of its operational capability these additional features:

- External control of tape speed, data mode, record, and playback functions.
- Status signals for "recording data", "reproducing data", or "recorder malfunction".
- Signals to telemetry for case temperature, internal pressure, and operating status.
- Internal power supplies and inverters for operation of the tape transport and DSA electronics from 28 volts (Nominal) supplied by the S/C Support

DSA BLOCK DIAGRAM

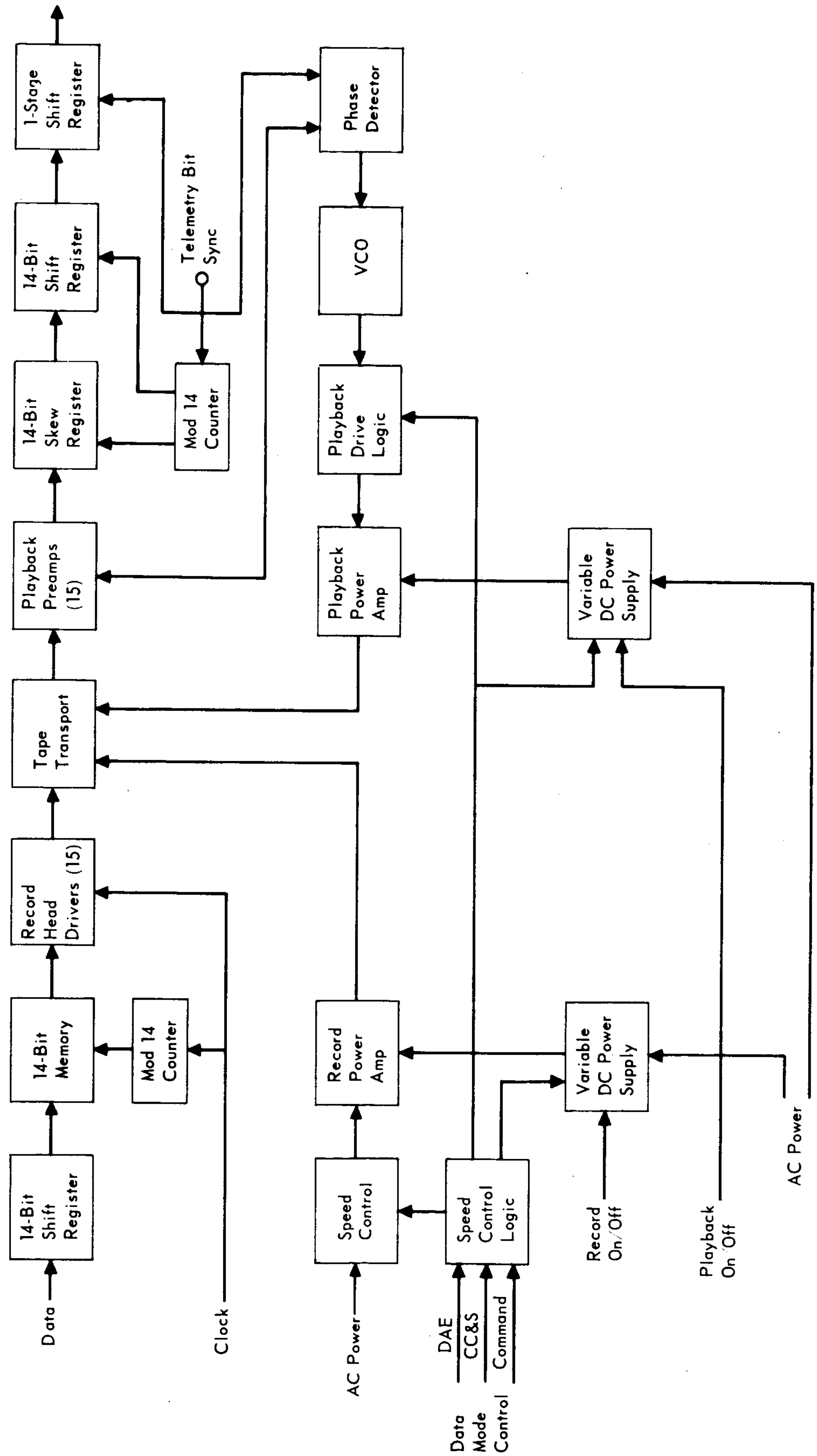


Figure 3.2-1

Electronics power bus.

3.2.5 Performance Characteristics - The performance characteristics of the DSA are given below:

Capacity:	30×10^6 bits
Data Input:	55,860 bits per second serial NRZ
Record Tape Speed:	3.99 inches per second (14 track parallel)
Bit Packing Density:	1000 bits per inch per track
Record/Reproduce Ratio:	7.76 to 1
Reproduce Tape Speed:	0.51 inches
Data Output:	7200 bits per second serial NRZ
Wow/flutter	Not applicable
Bit error rate:	Less than 1 bit in 10^5

3.2.6 Interface Definition - All DSA electrical interfaces except power and telemetry points are with the S/C-MTD Flight Capsule Data Distribution Unit (DDU); although, the sources and destinations of the electrical signals rest in other subsystems as listed in Figure 3.2-2.

The power interface is with the FSC Power Subsystem. The telemetry points interface with the S/C-MTD Support Equipment Instrumentation and Commutation Assembly, from which they are routed to FSC telemetry.

3.2.7 Reliability and Safety Considerations

3.2.7.1 Mission Success Definition - The Spacecraft Mounted Entry Science Package Data Storage (Tape Recorder) will record the high rate ESP data during the Descent and Terminal Descent Phases of the mission and play back the recorded data upon command from the Spacecraft. The record mode performance and the playback mode performance are compatible with the ESP data rates and FCS data rates, respectively, for mission success.

3.2.7.2 Reliability Model - The Tape Recorder (Data Storage) has an assessed reliability of a 0.986 probability of successfully performing its required function. Figure 3.2-3 illustrates the reliability model for the tape recorder.

3.2.7.3 Mission Failure Modes and Effects - Figure 3.2-4 depicts the failure fault tree for the tape recorder. Failure of the DSA in either the record or playback mode will result in a loss of the high rate entry science data (video) which results in a degraded mission. The low rate entry science data will not be lost due to redundant data transmission via the Capsule Bus link.

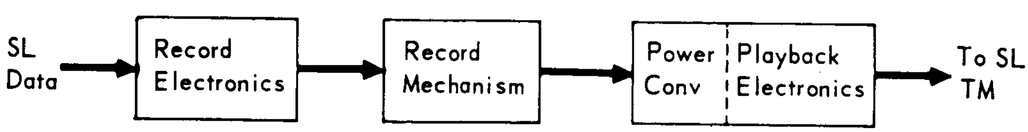
3.2.7.4 Complexity Estimate - The Spacecraft Mounted Entry Package Support Data Storage is estimated to have a total parts count of 1,248 items including

DSA/DDU INTERFACE DEFINITION

INTERFACE FUNCTION	SOURCE	DESTINATION
Record On/Off	SC-MTD ESP Radio Bit Synchronizer	
Input Data		
Input Clock		
Playback On/Off	FSC CC&S FSC Command	
Output Data		FSC Telemetry
Output Data Sync	FSC Telemetry	
Mode Control	FSC CC&S FSC Data Automation FSC Command	
Recorder Identification		FSC Telemetry
Status Signals		FSC Data Automation FSC Telemetry
Redundancy		
Reproducing		
Malfunction		
Start of Tape		
End of Tape		

Figure 3.2–2

SURFACE LABORATORY DATA STORAGE RELIABILITY MODEL



$$P_S(\text{Stored Data, 149 hr}) = e^{-0.014198} = 0.9859$$

Figure 3.2–3

SURFACE LABORATORY DATA STORAGE RELIABILITY FAULT TREE

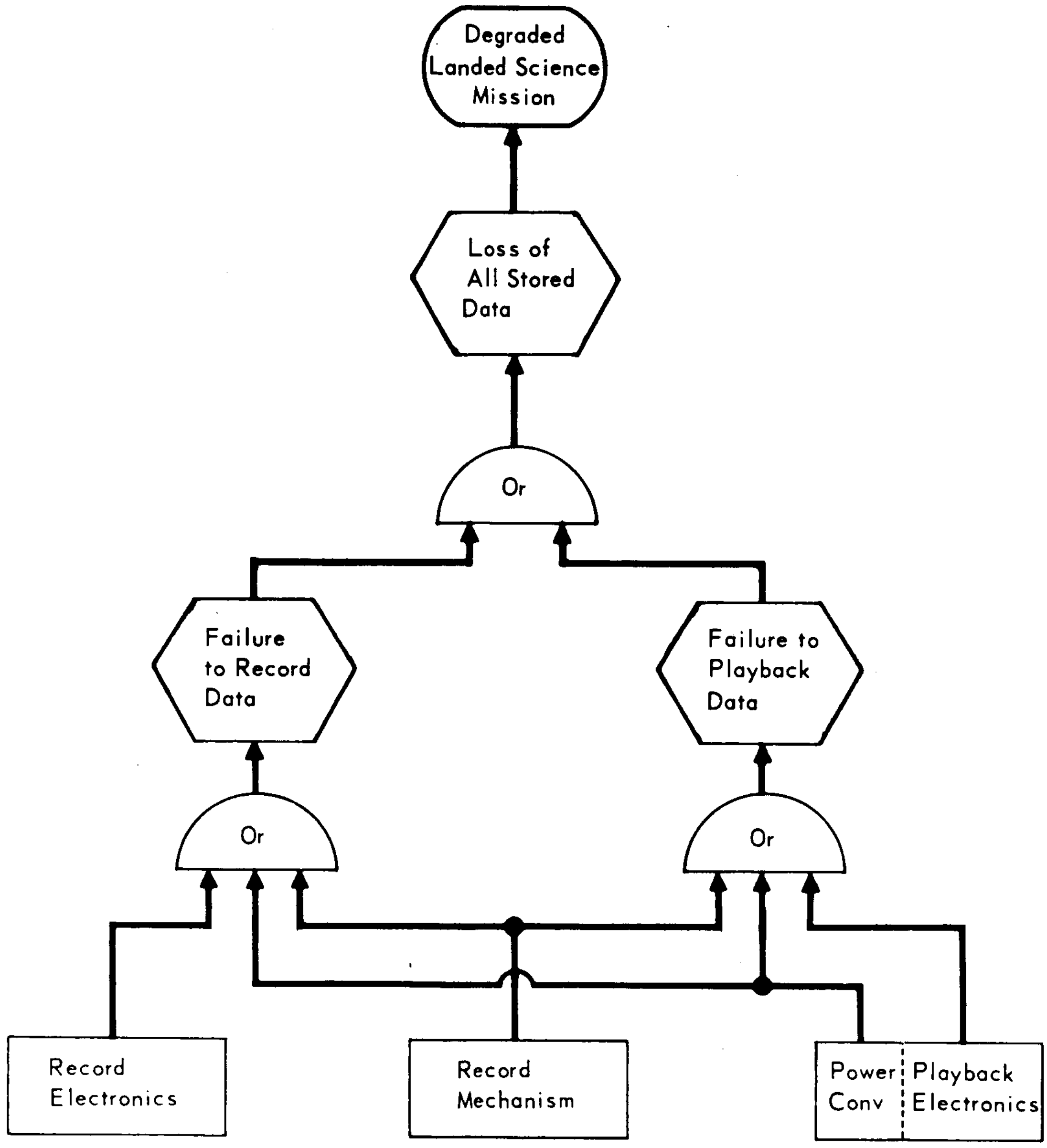


Figure 3.2-4

electronic, mechanical and electromechanical parts. Critical applications are bearings under shock and vibrational stresses. The parts count estimate is detailed in Figure 3.2-5.

3.2.7.5 Safety Considerations - Safety considerations are not applicable, since neither high voltage nor pyrotechnics are employed.

3.2.8 Test - The testing of the tape recorder is accomplished by recording a check stream on the tape and verifying the check words in the playback mode. Testing of the recorder is done as part of an end to end check of the S/C MTD ESP Support Equipment. Phase B studies have determined that testing will be conducted within the constraints delineated in Figure 3.2-6, in accordance with the Integrated Test Plan. Tests will be performed prior to canister enclosure, after enclosure prior to final sterilization, after sterilization and during pre-launch operations. All ground testing shall be performed with the use of Operational Support Equipment (OSE). In-flight checkout will be performed through the DSIF. In-flight checkout (pre-separation) is initiated by the Capsule Bus test programmer upon command from the DSIF and monitored through the spacecraft telemetry system.

3.2.9 Development Status - Tape blocking, tape-to-head adhesion and migration of bearing lubricant as a result of long periods of non-operation during the Interplanetary Cruise Phase will require additional study.

SURFACE LABORATORY DATA STORAGE RELIABILITY PARTS COUNT ESTIMATE

TAPE RECORDER

PART TYPE	λ	RECORD n	PLAYBACK n	TOTAL 2n	$\Sigma n\lambda$	RECORD MECHANISM PART TYPE	λ'	n	$n\lambda$
Capacitor									
Ceramic/Glass	11	14	121	125	125	Drive Belt	1000	4	4000
Solid Tantalum	2		50	50	100	Bearing Assembly	200	2	400
Diode						Pulley Shaft Assembly	1	2	2
General Purpose	0.1	28	166	194	213	Shaft Brg (Ball)	200	4	800
Zener/Refer.	10		11	11	110	Input Brg (Ball)	200	4	800
Resistor						Clutch Spring	250	2	700
Carbon Comp	0.1		2	2	0.2	Pulleys, Input	1	2	2
Film, Carbon/Metal	0.3	56	561	617	186	Tape Drive Belt	1000	1	1000
Variable	11.9		1	1	12	Springs	1	2	2
WW Power	0.5		2	2	1	Pivoted Sleeves	10	2	20
Potentiometer	108		1	1	108	Bearings	200	2	400
Transistor						Ball Brg Pair	200	1	200
General Purpose	5	18	168	186	930	Housing & Ctr Shift	1	1	1
Transformer	5	1	5	6	30	Hub Assy Rotating	1	1	1
Relay	10	1	1	2	20	Tape Roller Assemb	200	6	1200
Inductor	7.5		3	3	24	Mylar Mag. Tape	1	1	1
SCR	20		2	2	40				
Rectifier	120		2	2	240				
Record Head	20	2		2	40				
Playback Head	20		2	2	40				
Motor, Drive	300	1	2	3	900				
Totals	–	121	1100	1211	3119		–	37	9529
		(497)	(2611)		(3119)				(9529)

Figure 3.2-5

SURFACE LABORATORY SYSTEM
DATA STORAGE SUBSYSTEM
TEST MATRIX

27.0

In-Flight Checkout		Pre-Launch				System Test		Telemetry Points			TEST	ACCY REQ
											Non Operative Test	
						X					Input Bit Stream	Digital
						X					Output Bit Stream	Digital
		X				X		X			Memory "Overflow" Signal	Digital
		X				X		X			Memory "Empty" Signal	Digital
X		X				X		X			Power Supply Voltage	2%
X		X				X		X			Power Supply Current	2%
						X					Sequencing Clock Frequency	Digital
						X					Tape Speed	0.1%
X		X				X		X			Tape Direction	Digital
X		X				X		X			Tape Bias Voltage	2%
X		X				X		X			Tape Motor Drive Voltage	2%
X		X				X		X			Command Verification	Digital

Figure 3.2-6

SECTION 4

TELEMETRY SUBSYSTEM

4.1 EQUIPMENT IDENTIFICATION AND USAGE - The Telemetry (TM) Equipment on the Entry Science Package (ESP) has two functions. The first function is to format the ESP interplanetary cruise data under the control of the Capsule Bus (CB) Cruise Commutator, into a coherent bitstream and transfer this data to the CB cruise commutator for subsequent transmission by the Flight Spacecraft (FSC). The second function is to format the ESP entry data, and after proper interleaving with the Entry Science Package Data Storage Subsystem, transfer this bitstream to the ESP Radio Subsystem for transmission to the FSC.

The Cruise Commutator portion of the ESP TM has the following major components:

a. Analog gates (both single ended high level and double ended low level)
b. Differential amplifier
c. Analog to digital converter (including a sample and hold)
d. Digital multiplexer (including Schmidt triggers for non-logic level bilevel inputs, and "holding" circuitry for pulsed bilevel inputs).

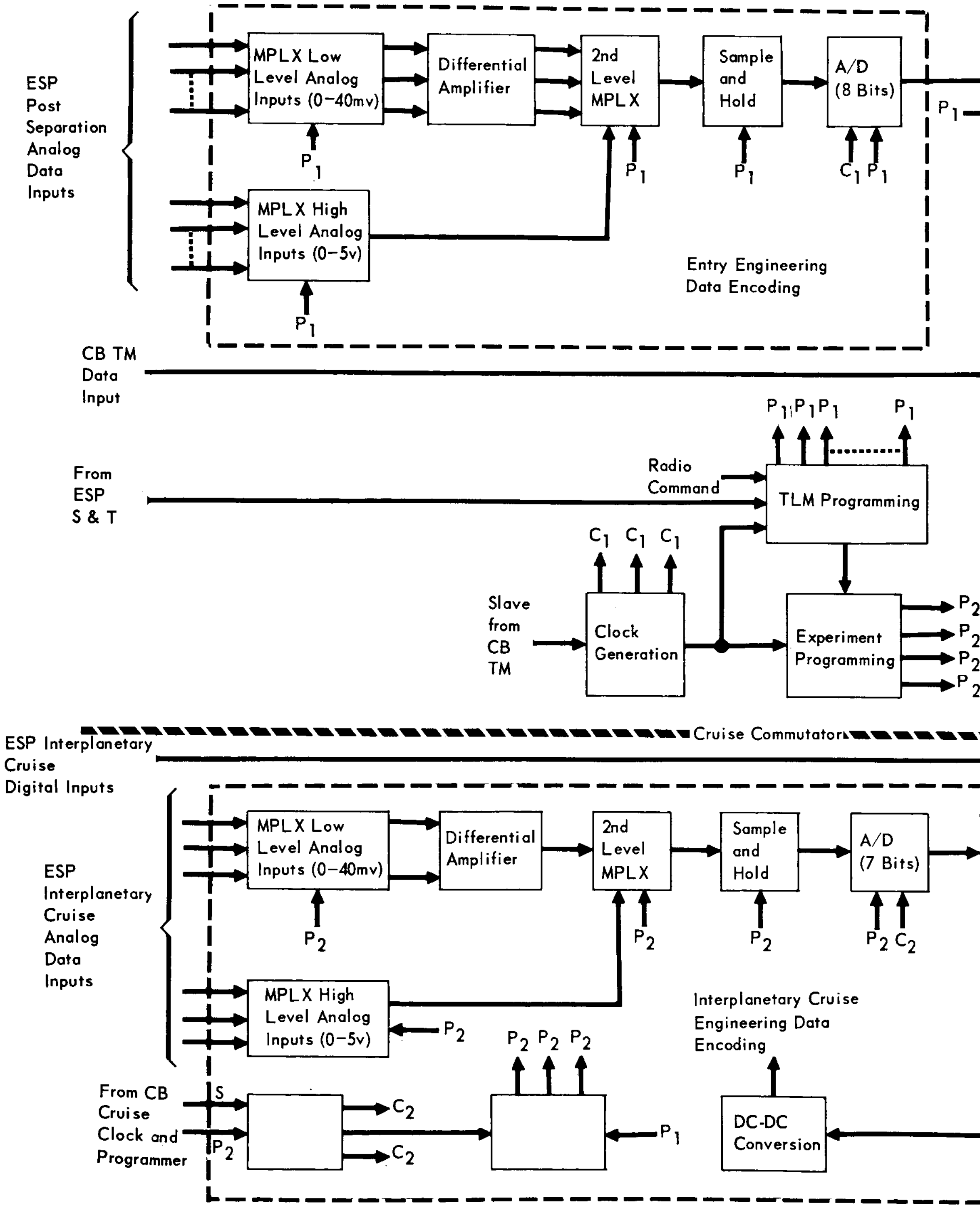
The Main Commutator portion of the ESP TM has the following major components:

a. Analog gates (both single ended high level and double ended low level)
b. Differential amplifier
c. Analog to digital converter (including a sample and hold)
d. Digital multiplexer (including Schmidt triggers for non-logic level bilevel inputs, and "holding" circuitry for pulsed bilevel inputs).
e. Frame and subframe synchronization generators
f. Clock
g. Sequence programmer for data formating and experiment control
h. Dc to dc converter
i. Bit, word, and frame interleavers
j. Biphase (Manchester II + 180) encoder

Figure 4-1 is a functional block diagram of the ESP TM. Figure 4-2 illustrates the usage of the ESP TM in terms of mission phases.

The data requirements of the ESP TM are defined in Figure 4.3-2, Part E, Volume IV, "The Entry Science Package Instrumentation List". This equipment works principally in conjunction with a) the Instrumentation Subsystem which provides the engineering

ENTRY SCIENCE PACKAGE TELEMETRY EQUIPMENT FUNCTIONAL BLOCK DIA



4-2-1

GRAM

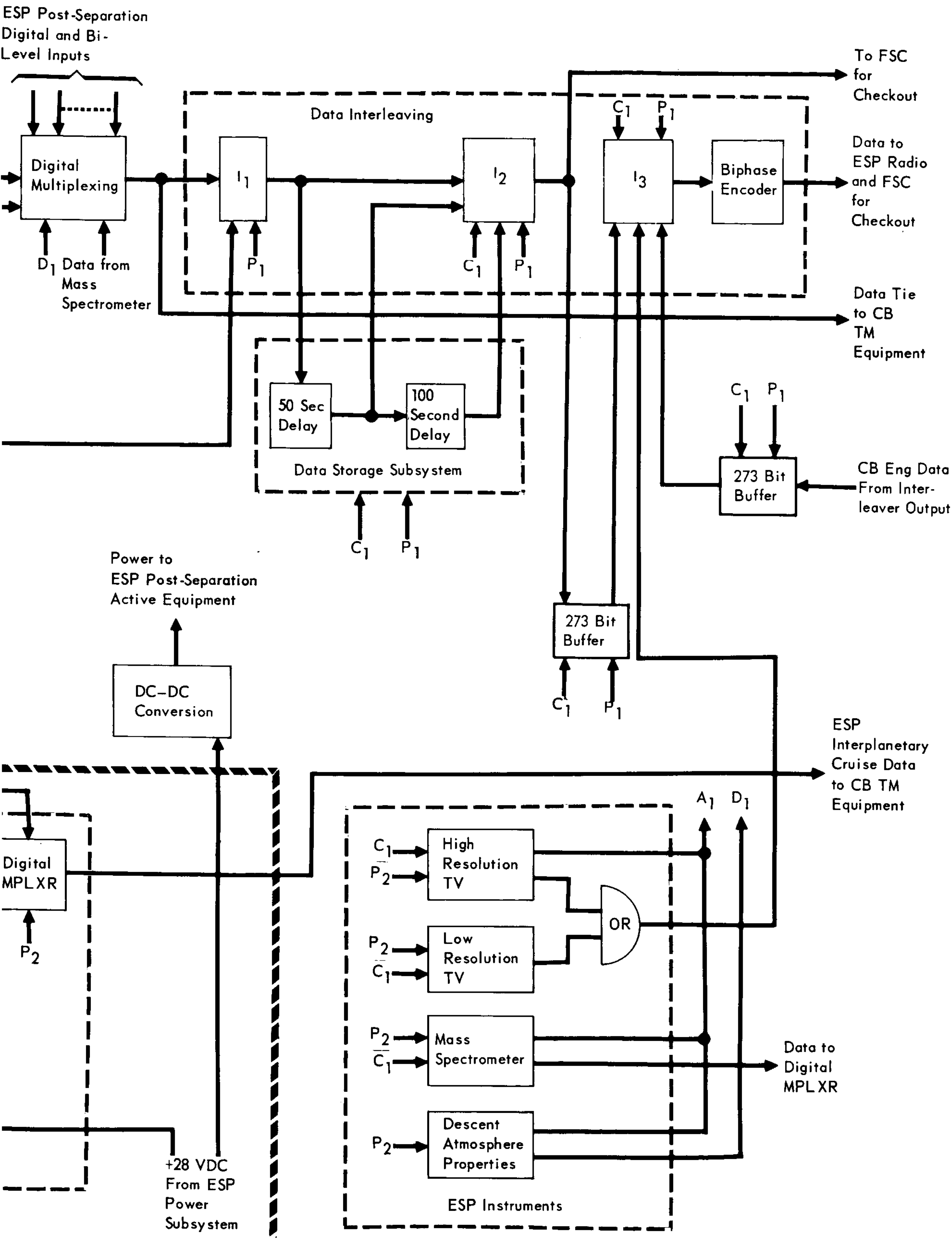


Figure 4–1

ENTRY SCIENCE PACKAGE ACTIVITIES AS RELATED TO MISSION PHASE

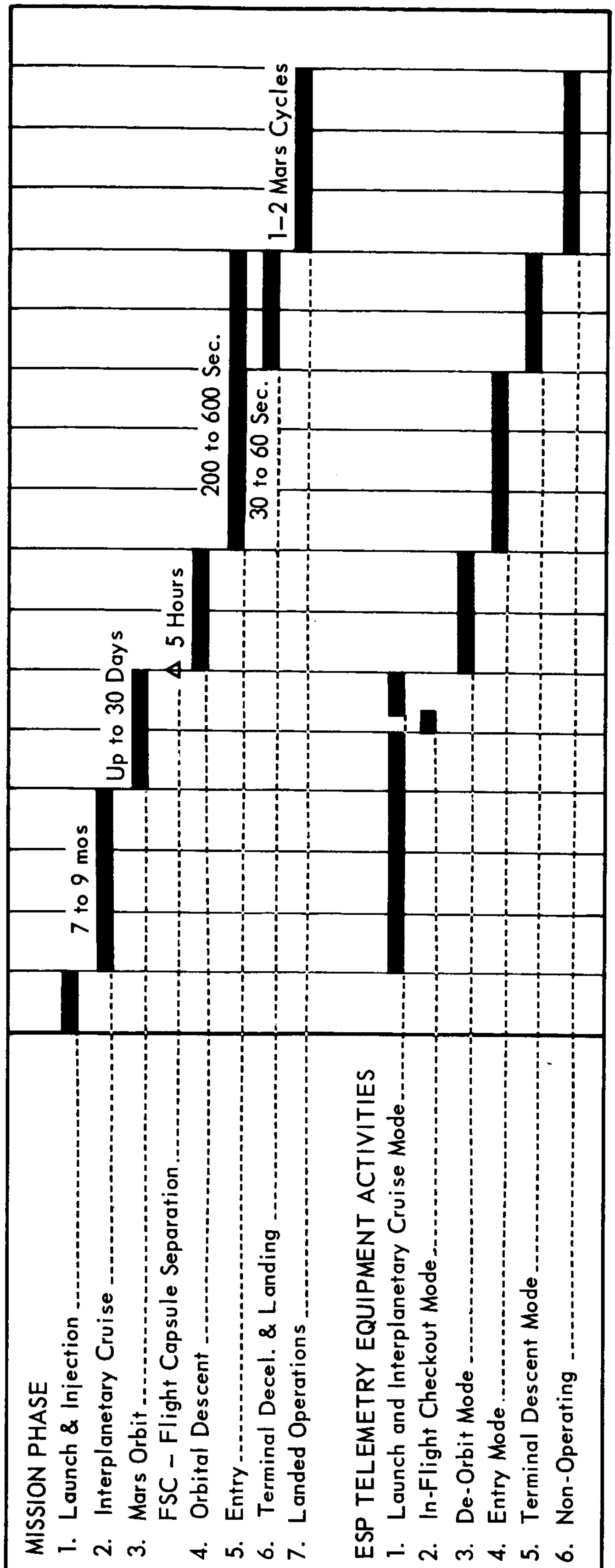


Figure 4–2

sensors (and their associated instrumentation power supplies) and processes much of the non-standard signals into standard signals, b) the ESP Radio Subsystem, c) the Spacecraft Mounted Data Distribution Unit, d) the CB Sequencer and Timer (ST) which initiates mode changes, e) the ESP Data Storage Subsystem, f) the CB Power Distribution Subsystem, and g) entry science instruments.

The ESP TM has a total of 86 signal inputs; 34 of which are single ended high level, 32 are double ended low level, 7 are bilevel and 13 are digital. These signals are arranged into 4 modes of operation, i.e., only those signals which are active during a given mission phase are telemetered.

4.2 DESIGN REQUIREMENTS AND CONSTRAINTS - The following usual VOYAGER requirements and constraints are applicable:

a. Sterilization
b. Low power consumption
c. Hard vacuum space environment
d. Life
e. Reliability
f. Satisfactory operation after a long dormancy.

The ESP TM also has the following specific data oriented requirements and constraints:

a. Reprogrammable formatting and experiment sequencing for mission flexibility
b. Significant redundancy in the Cruise Commutator - due to the long operating life
c. DC isolation of all digital interfaces reducing ground loop vehicle noise
d. Usage of alternate functional paths as opposed to block redundancy
e. Graceful degradation

4.3 PHYSICAL CHARACTERISTICS - The ESP TM occupies 240 cubic inches, weighs 9 lbs. and requires 5 watts.

4.4 OPERATIONAL DESCRIPTION - The operational description is divided into two parts. The first part describes the modes of operation of the equipment in terms of the mission phase. The second part describes the functions of the principal components.

4.4.1 Operational Modes - The ESP will operate in four modes: launch and interplanetary cruise, in-flight checkout, deorbit, and atmospheric entry terminal descent.

4.4.1.1 Launch and Interplanetary Cruise Mode - During this mode the bulk of the ESP equipment is dormant, thus only a small amount of status information is required. Only the Cruise Commutator will be active during this mode. The ESP

Cruise Commutator will multiplex the cruise data and transfer a single data stream to the CB Cruise Commutator for subsequent transfer to the Data Distribution Unit in the spacecraft. In this mode the Cruise Commutator will be subject to the control (both bit, word, and frame slave) of the CB Cruise Commutator. The data format for this mode is given in Figure 4-3. These data are transferred at 0.4375 bps rate.

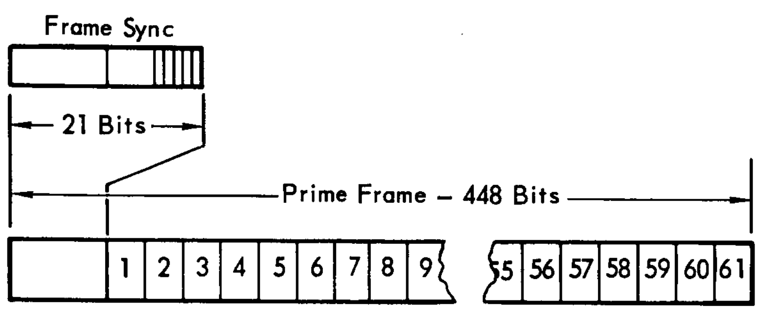
4.4.1.2 <u>In-Flight Checkout Mode</u> - During this mode all of the ESP subsystems are tested prior to separation. The ESP TM will have three "real" modes here; a memory dump, and two modes to support the checkout of the ESP subsystems. A fourth "quasi-mode" will consist of cycling the ESP TM through all of its modes. During the normal checkout mode, the Cruise Commutator is read through the main CB TM. The formats for the two checkout modes are given in Figures 4-4 and 4-5. The Entry checkout data are transferred to the Data Distribution Unit storage at 55,860 bps and played back at 2730 to the FSC TM. The engineering and low rate science data are transferred to the FSC TM at a 2730 bps rate.

4.4.1.3 <u>Deorbit Mode</u> - The deorbit mode covers the mission phases from the separation of the CB from the FSC to entry (at approximately 800,000 feet). In this mode the ESP Cruise Commutator functions as a remote multiplexer to the CB Cruise Commutator. The combined ESP and CB deorbit format is given in Volume VI, Part C, Section 2.1.

4.4.1.4 <u>Entry/Terminal Descent Mode</u> - From 800,000 feet to approximately 5 minutes after touchdown the TM is in the entry/terminal descent mode. The format for this mode is shown in Figure 4-6. During atmospheric entry there is a high probability of shock induced ionization blackout, thus the non-television data are sent in both real time and delayed time. During this time the main CB TM is interleaved with the ESP TM. The combined delayed and real time CB TM and ESP non-television is then combined with the television data. The format for this frame interleaving is shown in Figure 4-7. This bitstream at 55,860 bps is biphase (Manchester II + 180) encoded and transmitted through the ESP Radio Subsystem.

4.4.2 <u>Component Description</u> - The "commutation" portions of the Main and Cruise Commutators are functionally identical and similar to conventional telemeters; i.e., the analog gating, differential amplifier, sample and hold, analog to digital conversion, digital multiplexing and synchronizing operations. The "programming" portions of the two commutations are distinctly different. The Cruise Commutator is a hardwired device, employing conventional matrix programming. The Main Commutator is a stored program device, employing interlaced tube programming. The data storage interlacing operation is unique to the main TM. The following component description is divided into three parts; commutation, programming, and interleaving.

ENTRY SCIENCE PACKAGE
LAUNCH AND INTERPLANETARY
CRUISE MODE TELEMETRY FORMAT.



CHANNEL (WORD) DESIGNATIONS	SAMPLE RATE*	CHANNELS (WORDS) AVAILABLE	CHANNELS USED
1 → 21	.001	61	21

* Based on a bit rate of .4375 bits per second, and a sample word size of 7 bits.

Figure 4-3

ENTRY SCIENCE PACKAGE IN-FLIGHT CHECKOUT MODE TELEMETRY DATA FORMAT

NO. OF CHANNELS	DATA TYPE	SAMPLE RATE (sps)
1	BL	1
3	HL	.1
5	LL	.1
9	HL	.05
2	LL	.05
10	HL	1/30 = .03
5	LL	1/30 = .03
1	BL	1/30 = .03
7	D	1/30 = .03
12	HL	.01
23	LL	.01
1	BL	.01
4	D	.01
1	D	8

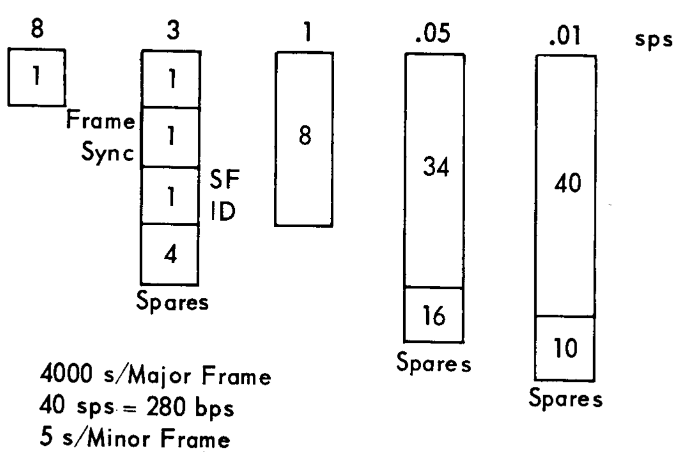


Figure 4-4

ENTRY SCIENCE PACKAGE IN-FLIGHT CHECKOUT MODE TM FORMAT FOR VIDEO DATA

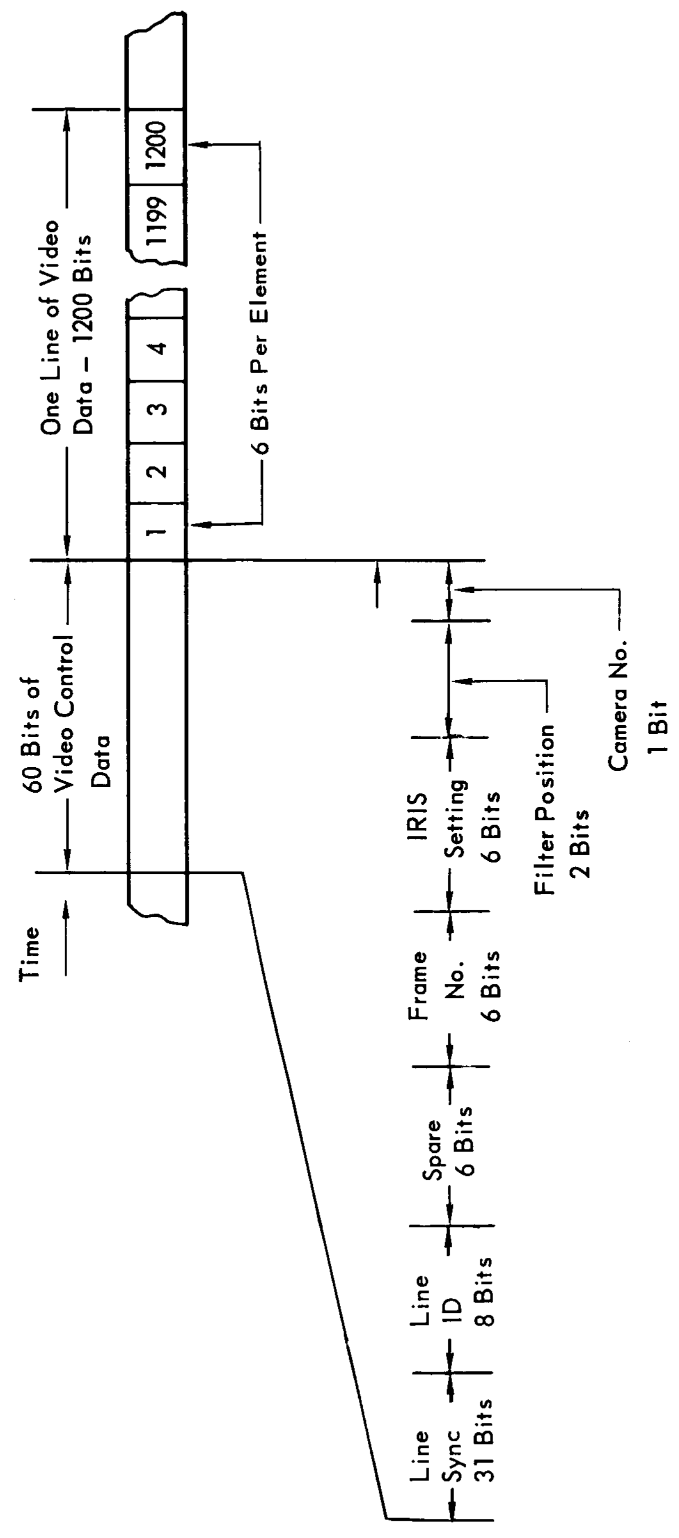


Figure 4-5

ENTRY SCIENCE PACKAGE TELEMETRY EQUIPMENT

ENTRY AND TERMINAL DESCENT TM FORMAT

NUMBER OF CHANNELS	DATA TYPE	SAMPLE RATE (SPS)
1	D	8
4	HL	1
4	LL	1
1	BL	1
1	BL	2
6	D	2
8	HL	.1
6	LL	.1
1	D	.1
8	HL	.05
7	LL	.05
1	BL	.05
2	D	.05
5	HL	.01
13	LL	.01
1	D	.01

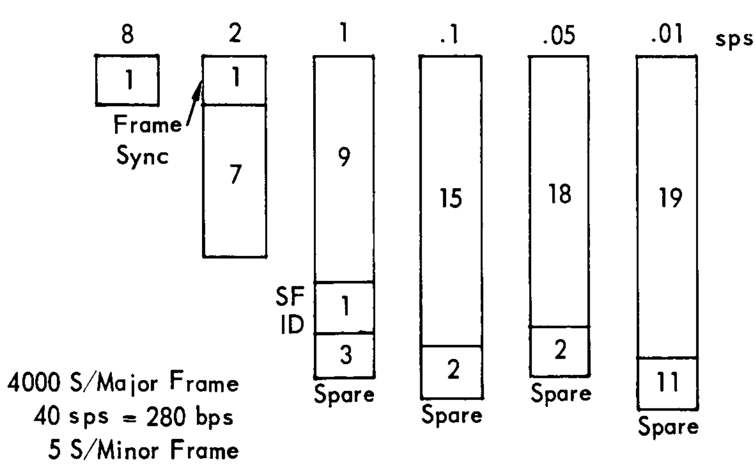


Figure 4-6

ENTRY SCIENCE PACKAGE DATA INTERLEAVING TIMING

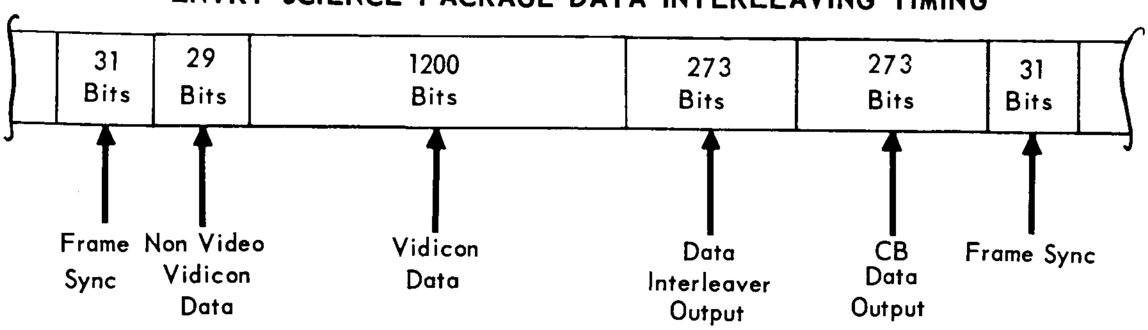


Figure 4–7

4.4.2.1 <u>Commutation</u> - The input signals are gated through MOSFET switches appropriately "treed"; that is, the gates are arranged into subgroups so that the failure of a single input switch will not propagate the failure any further than the subgroup. The specific treeing design is a function of the maximum allowable backcurrent into the data sources, programming efficiency and, most importantly, reliability. The single ended high level signals are gated directly to the sample and hold portion of the analog to digital converter. The double ended low level signals first pass through a differential amplifier, which converts them to single ended high level signals, and then to the sample and hold. The sample and hold charge time is chosen to minimize aperture errors. The analog to digital converter output is gated through the digital multiplexer together with the frame and subframe synchronization words and the bilevel data. All of the bilevel data is buffered - the "logic level" digital data being directly buffered, while the "non-logic" level bilevels are passed through Schmidt triggers for voltage conversion. Some of the bilevels are pulsed, thus require "holding" circuitry. Dependent upon the specific accuracy requirement the bilevels may or may not be time tagged. The output of the digital multiplexer is the coherent PCM bitstream. A 10% spare channel capability has been incorporated into the commutator.

4.4.2.2 <u>Programmer</u> - The ESP Cruise Commutator programming is derived directly from the CB Cruise Commutator; i.e., it functions as a true remote multiplexer and thus has no internal programming.

The main ESP TM programmer is a stored program device. The telemetry formatting employs interlaced tube programming technique, while the experiment sequencing is normal random access programming. A block diagram of the programmer is shown in Figure 4-8. The magnetic core memory utilizes 20 mil cores in a 3D organization with a total of 8192 bits. The clock herein drives a "hardwired" tube structure for the TM formating - any given position in the tube being a switch identified in core.

The interlaced tube memory has a unique address for each data channel. The "hardwired" tube structure is used to sequence the memory. The addresses of all the data channels are stored sequentially in memory. When a frame is to be sampled, the control unit accesses the memory, decodes the word, samples the channel, and adds one to the memory position for the next memory word. This process repeats until a rate group is completed. When a rate group is completed, and it is not time to repeat the group, the next lowest rate group is initiated until it is time

ESP TM EQUIPMENT PROGRAMMER

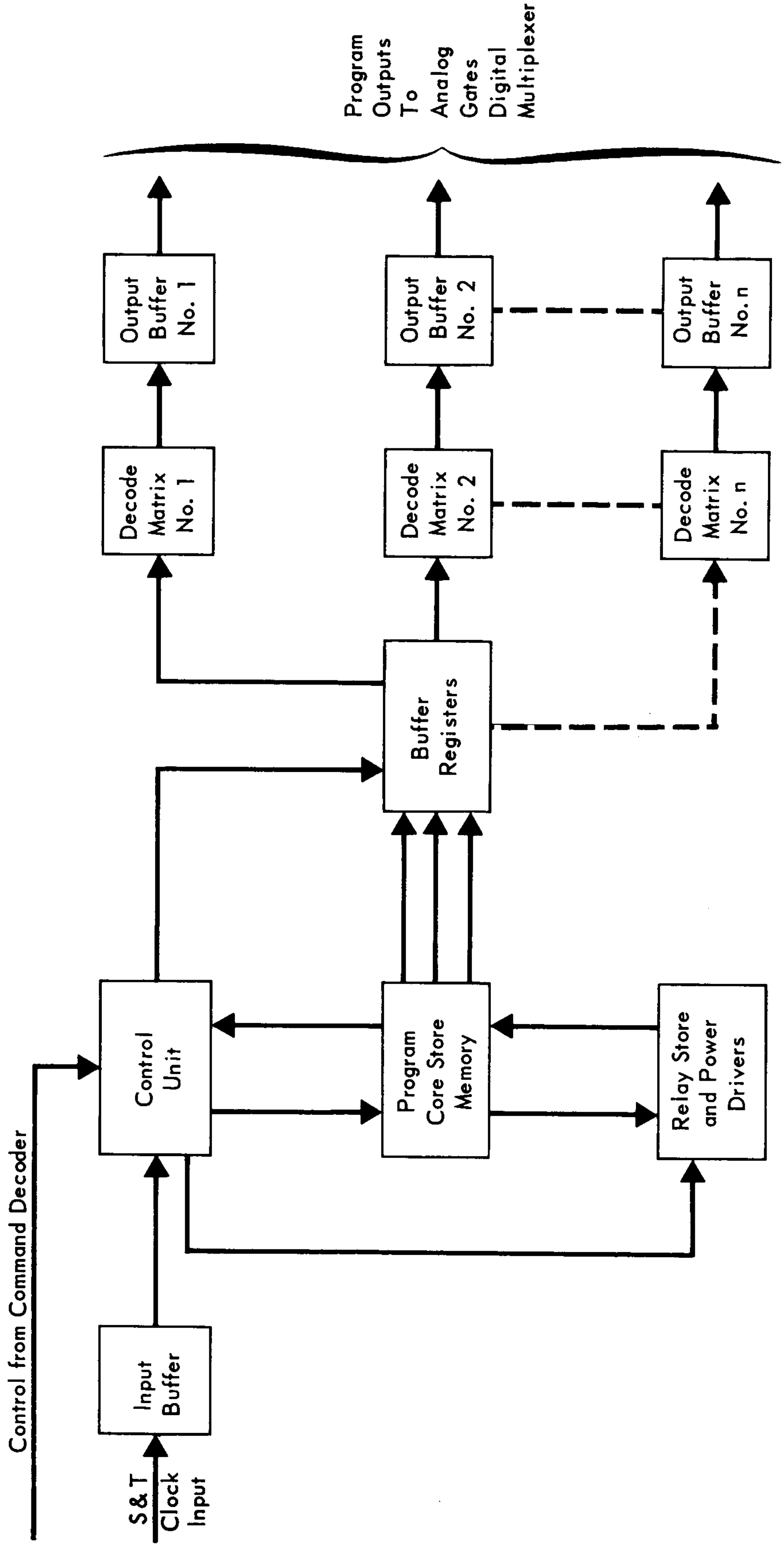


Figure 4-8

to reinitiate the sampling of the next highest group.

The combination of a hardwired logic tube structure and stored program switch position assures a) that a single core failure will not disrupt the entire format, and b) a new switch position may be easily reprogrammed into core by a simple core address technique.

Part E, Section 4, contains a definitive description of the interlaced tube programming technique.

4.4.2.3 Interleaver - The ESP interleaving is shown in Figure 4-1. In the entry/terminal descent modes the main CB TM is interleaved with the ESP main TM on a word basis. The ESP rate is 280 bps and the CB rate is 630 bps. Next this bitstream is interleaved on a bit basis with the 50 and 150 second delayed streams from the ESP Data Storage Subsystem to form the composite. This composite stream at 2730 bps is then interleaved with a 2730 bps CB bitstream, and the television output resulting in a 55,860 bps rate. Figure 4-9 illustrates the mechanics of this final interleaving process. The 55,860 bps steeam is biphase (Manchester II + 180) encoded prior to transmission by the ESP Radio Subsystem.

4.5 PERFORMANCE CHARACTERISTICS

Input Signals

a. Single ended high level, 0-5V, 0-5 kilohm
b. Double ended low level, 0-40 V, 0-500 ohm, maximum of 10V common mode
c. Logic level digital inputs - 0 or 5V, 0-5 kilohm
d. Nonlogic level bilevel - 0 or 28V, 0-10 kilohm

Conversion Accuracy

a. Single ended high level; $\pm$ 1 count in 126 counts - for the Cruise Commutator and $\pm$ 1 counts in 254 or 126 counts in the Main Commutator as a function of the signal accuracy requirement.
b. Double ended low level; $\pm$ 2 counts in 126 counts - for the Cruise Commutator and $\pm$ 1 count in 254 or 126 counts in the Main Commutator as a function of signal accuracy requirement.
c. Logic level digital, 1 error in 10^5
d. Non-logic level bilevel, 1 error in 10^3 with less than 1V as a "space" and greater than 4V as a "mark".

Output Signals

a. All digital, 0 or 5V, 5 kilohm
b. Straight binary encoded NRZ zero and full scale suppression out of ADC

ESP TM EQUIPMENT DATA INTERLEAVER NO. 3 LOGIC

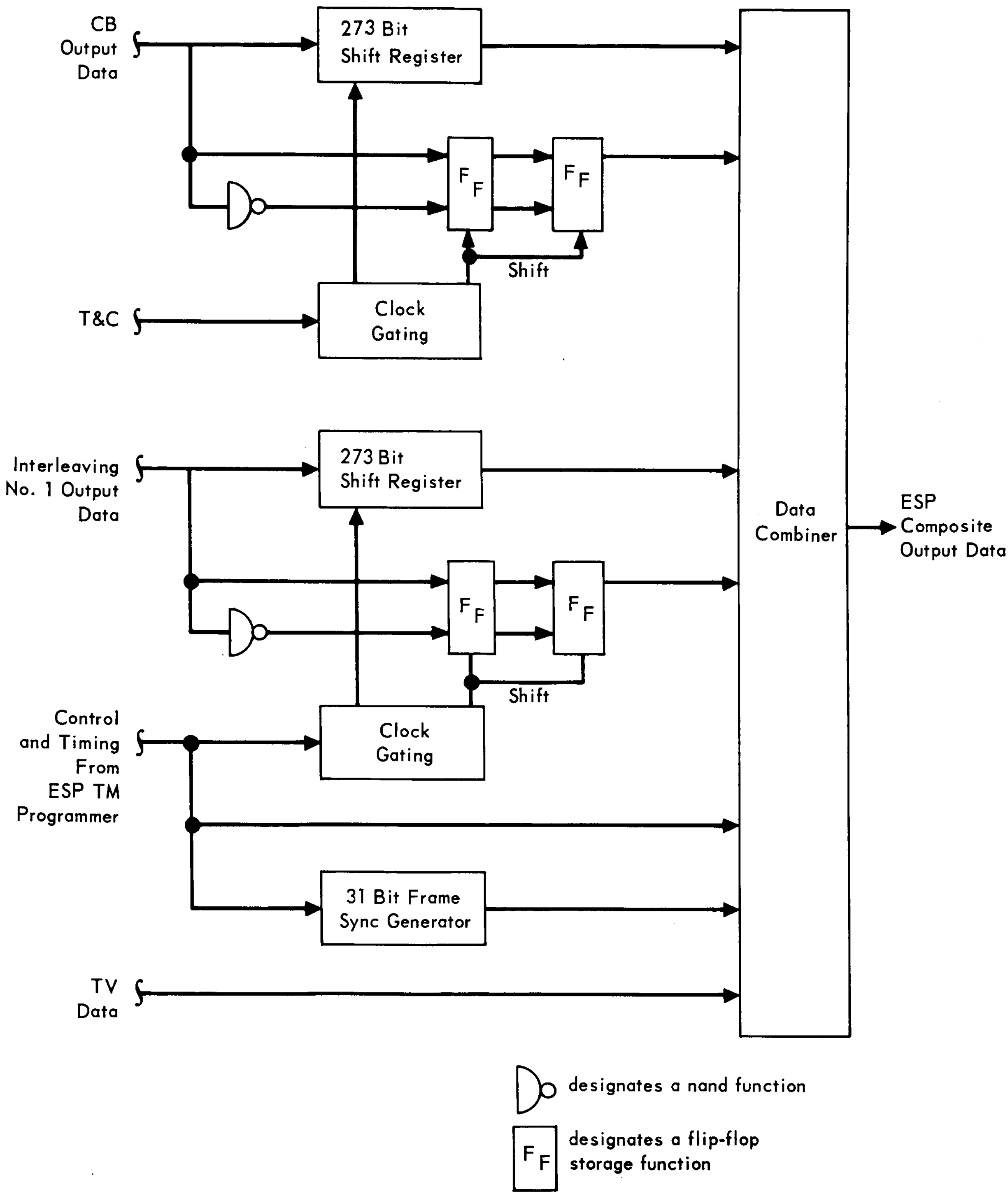


Figure 4-9

c. Biphase level coded serial PCM from main TM-NRZ coded serial PCM from Cruise Commutator.

Programming

a. Interlaced or burst tube programming for all stored programs

b. Matrix or interlaced tube programming for all hardwire programs

c. Reprogram switch positions by radio command in stored programs

d. Random access all memories

General

a. Analog switching action is independent of source impedance

b. DC isolation in all digital interfaces

c. Signal, power, and chassis ground isolation

d. Vehicle time in each frame

4.6 INTERFACES - The interfaces are given in Figure 4-10.

4.7 RELIABILITY AND SAFETY CONSIDERATIONS - The ESP Telemetry Subsystem reliability mode and fault tree are shown in Figures 4-11 and 4-12. The estimated reliability factor for the ESP Telemetry Subsystem is .987. The Cruise Commutator and Encoder has an estimated reliability of .998, excluding programming functions. Due to the continuous operation of the Cruise Commutator and Encoder through interplanetary cruise, we have utilized active redundant analog data switches and a standby redundant analog-to-digital converter which is switchable by MOS command. Program function redundancy considerations require further analysis. Alternatives include operation on a duty cycle basis, additional circuit redundancies or decentralization programmer timing functions to permit limited operation and data retrieval in spite of individual failures. No special safety provisions are required because there are no high voltage applications in the Telemetry Subsystem.

4.8 TEST - The bulk of the Telemetry using subsystems are tested through the telemetry subsystems during the test build-up from factory test through lift-off. During the Mars orbit in-flight checkout mode, all of the using subsystems are tested through the Telemetry Subsystem. Prior to any of the using subsystems tests the Telemetry is tested. The telemetry tests are conducted in three steps, a memory readout mode, a "cycle" through all modes, and a checkout mode. (See Figure 4-13). The telemetry tests are principally directed toward the major functional blocks by following a specific data "chain". Each "chain" is identified by a block of signal types; e.g., a low level calibration signal(s) is injected to a low level gate and the output of the TM (at the Radio Subsystem input is monitored). This tests

ESP TELEMETRY INTERFACES

	Instrumentation Equip	S and T	ESP Power Bus	ESP Radio	ESP DSS	FSC Command	DDU	CB Cruise Digital Mux	CB Cruise Programmer	Science Instruments
Main ESP Telemetry										
Analog Gates	X									X
Digital Multiplexer	X	X								X
Data Interleaver				X	X		X			
Clock		X								X
Programmer		X				X				X
DC to DC Converter			X							
Cruise Commutator										
Analog Gates	X									
Digital Multiplexer	X	X					X	X		
Gate Drivers									X	
DC to DC Converter			X							

Figure 4–10

ENTRY SCIENCE PACKAGE
TELEMETRY EQUIPMENT RELIABILITY MODEL

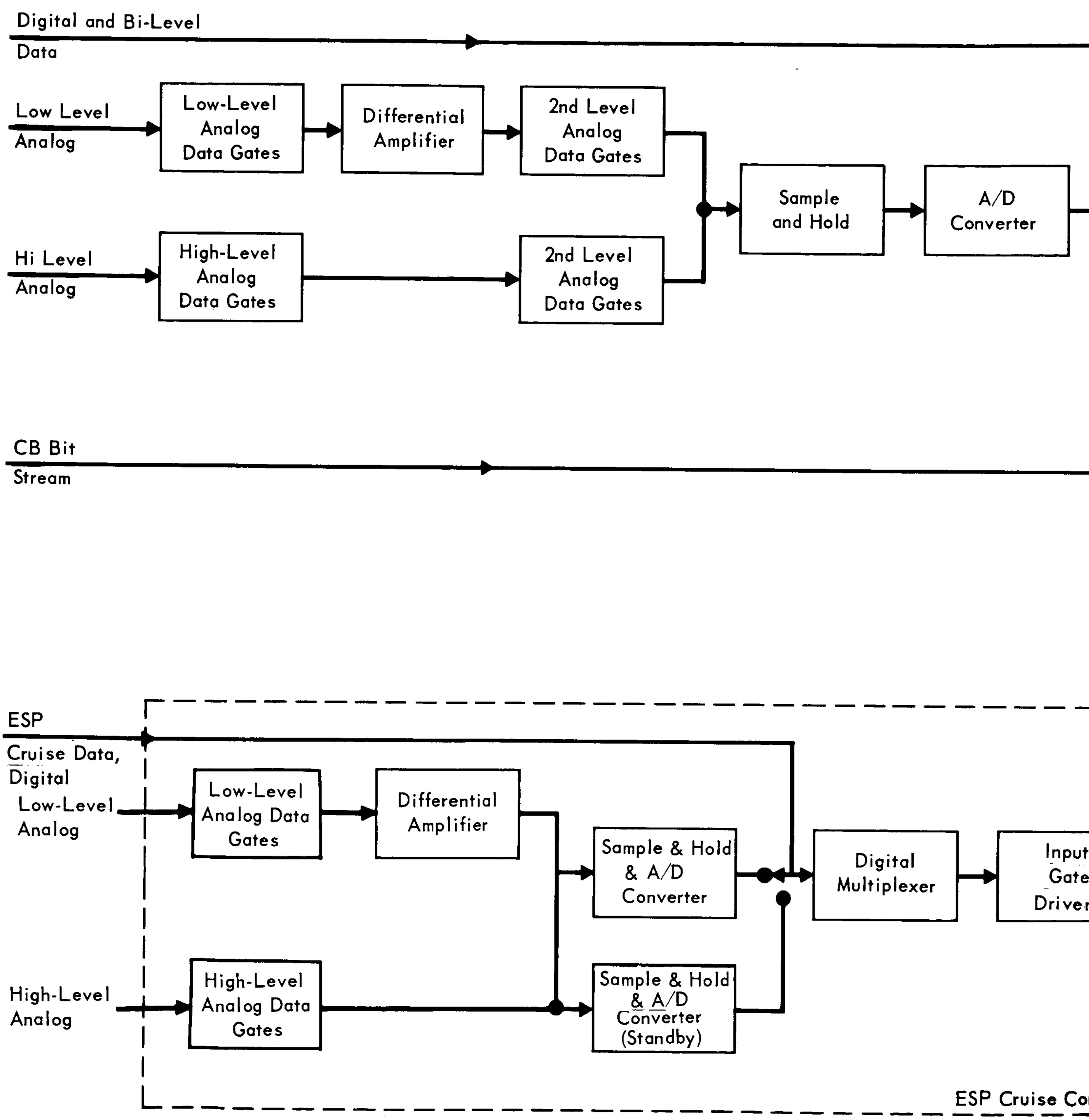
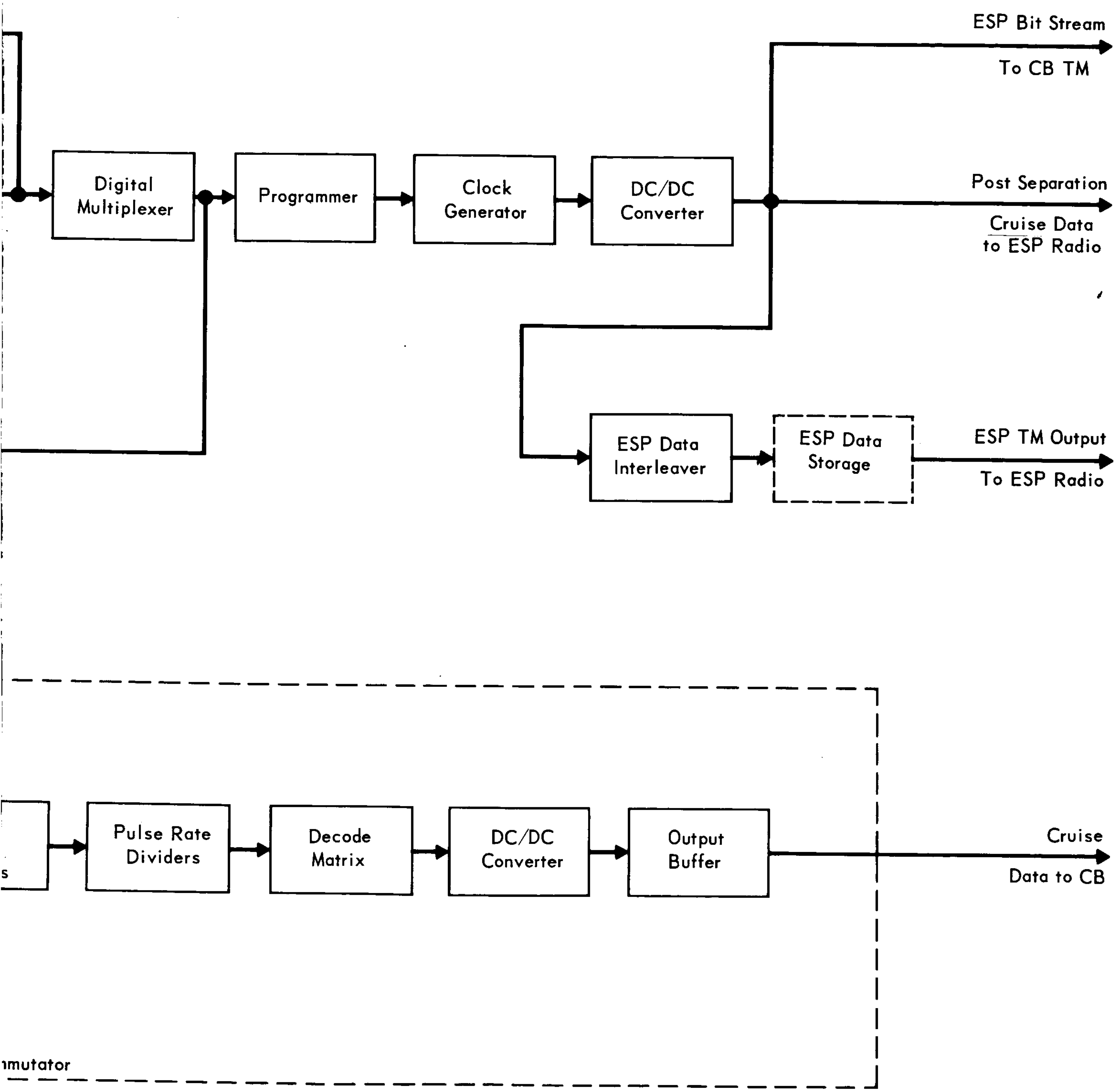


Figure 4-11


ESP Bit Stream
To CB TM
Digital Multiplexer
Programmer
Clock Generator
DC/DC Converter
Post Separation
Cruise Data to ESP Radio
ESP Data Interleaver
ESP Data Storage
ESP TM Output
To ESP Radio
s
Pulse Rate Dividers
Decode Matrix
DC/DC Converter
Output Buffer
Cruise
Data to CB
ımutator

VOYAGER ENTRY SCIENCE PACKAGE
TELEMETRY EQUIPMENT RELIABILITY FAULT TREE

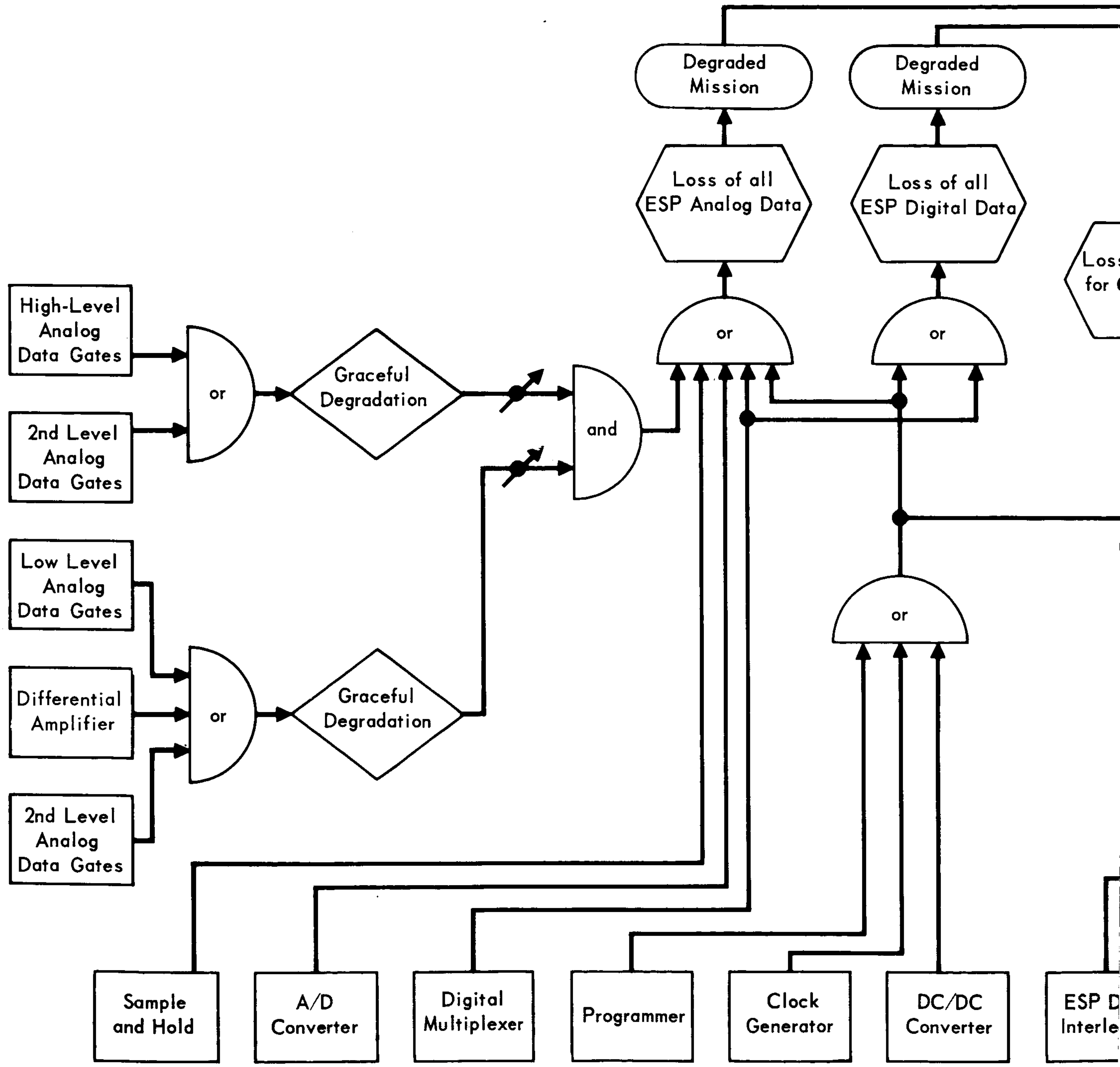
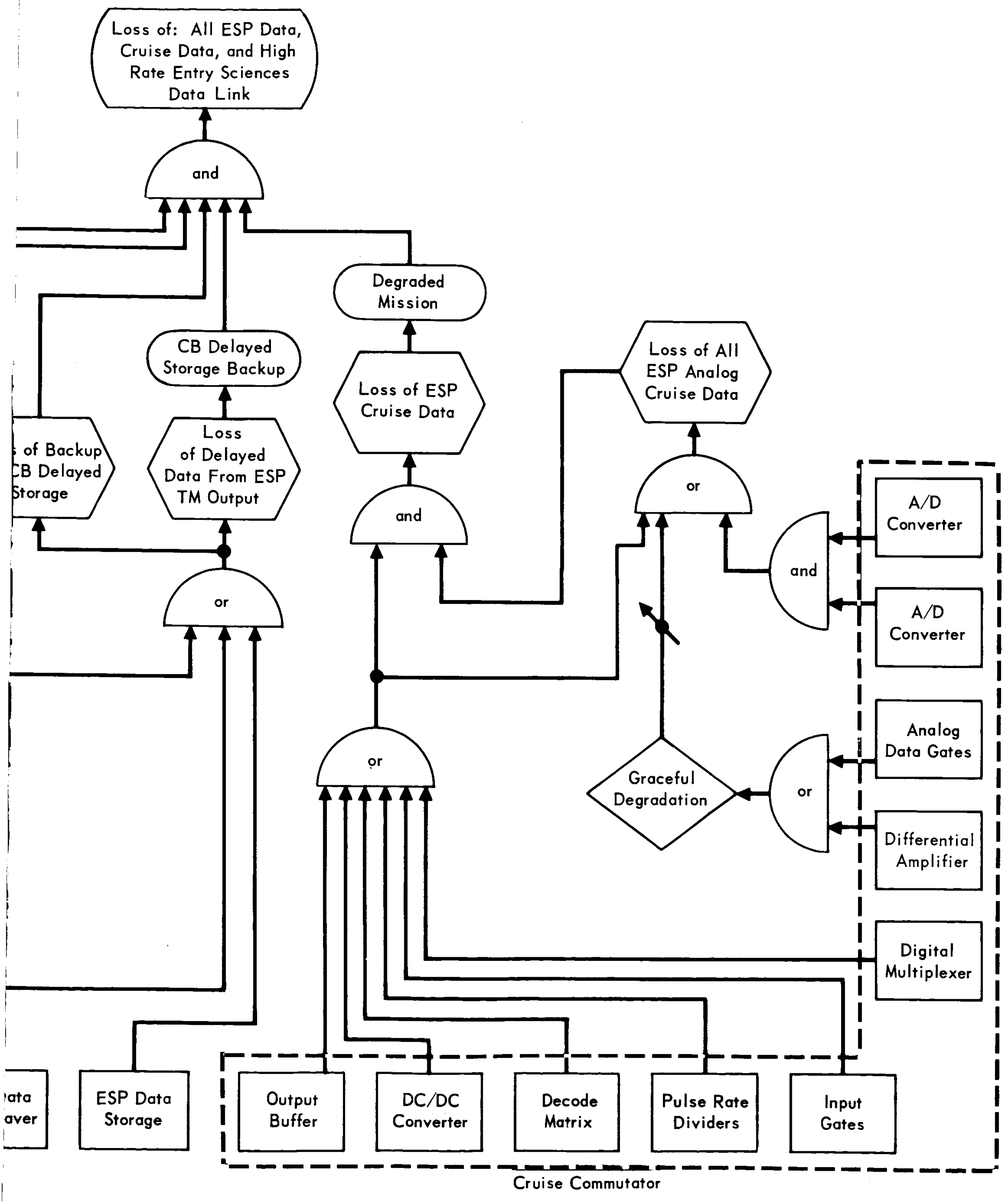


Figure 4–12


Loss of: All ESP Data, Cruise Data, and High Rate Entry Sciences Data Link
and
Degraded Mission
CB Delayed Storage Backup
Loss of ESP Cruise Data
Loss of All ESP Analog Cruise Data
of Backup
CB Delayed
Storage
Loss of Delayed Data From ESP TM Output
and
or
A/D Converter
and
or
A/D Converter
Analog Data Gates
or
Graceful Degradation
or
Differential Amplifier
Digital Multiplexer
ata
aver
ESP Data Storage
Output Buffer
DC/DC Converter
Decode Matrix
Pulse Rate Dividers
Input Gates
Cruise Commutator

4-18-2

ENTRY SCIENCE PACKAGE TELEMETRY EQUIPMENT TEST MATRIX

TEST	Non Operative Test	Multiplexer/ADC	Input Calibration Signals	A/D Calibration Signals	ESP Programmer	Input Commands	Output Gating Signals	Clock Generator	CB Control Input	Digital Multiplexer	Input Data Digital Test Pattern	Operative Sequence	Cruise Commutator	Input Calibration Signals	A/D Calibration Signals	Data Interleaver	Input Data	Control Input	Output Data	DC–DC Converter	Output Voltage
Accy Req			1%	1%		Digital	Digital		Digital		Digital	Digital		1%	1.0%						
Telemetry Monitor			X	X		X	X		X		X	X		X	X			X			X
System Test (Pre-Canister)			X	X		X	X		X		X	X		X	X		X	X	X		X
System Test (With Canister)			X	X		X	X		X		X	X		X	X			X	X		X
Pre-Launch			X	X		X	X		X		X	X		X	X			X	X		X
In-Flight Checkout			X	X		X	X		X		X	X		X	X			X	X		X

Figure 4–13

the low level signal equipment "chain"; namely, the gates and gate driver programming, the differential amplifier, the digital multiplexer, and the interleaver. Note that each individual gate is not tested (in-flight) but all of the major functional blocks are tested. Factory testing tests each gate, while the individual gates are tested in flight when monitoring the using subsystems.

4.9 DEVELOPMENT STATUS - Aside from the normal VOYAGER development requirements, specifically reliability and sterilization, the ESP TM will generally not require any state-of-the-art advances. The exception is the core stack drivers at low temperature. This problem in past space programs has been alleviated by close temperature control, however the battery weight necessary for the VOYAGER mission makes this solution prohibitively heavy.

SECTION 5

ENTRY SCIENCE PACKAGE RADIO SUBSYSTEM

The Entry Science Package Radio Subsystem consists of the Entry Science Package Radio (below) and the Spacecraft-Mounted Entry Science Package Support Radio (Paragraph 5.2). The radios are mutually compatible parts of the Entry Science Package to Spacecraft Relay Link, and they interface with each other through the Entry Science Package Antenna Subsystem.

5.1 ENTRY SCIENCE PACKAGE RADIO

5.1.1 Equipment Identification and Usage - The Entry Science Package (ESP) Radio is the transmitter portion of the Entry Science Package to Spacecraft relay link. The ESP transmitter is activated when the Flight Capsule enters the Mars atmosphere. Both television and engineering data will be transmitted during the period between entry and landing as a single PCM bit stream. The bit rate is 55,860 bps. The preferred radio consists of the following subassemblies:

a. 401.5 MHz Transmitter
b. 401.5 MHz Transmitter Power Converter
c. 341.5 MHz Transmitter
d. 341.5 MHz Transmitter Power Converter
e. Antenna Diplexer

A block diagram of the ESP Radio is shown in Figure 5.1-1. Both transmitters operate simultaneously to provide frequency diversity.

ESP RADIO

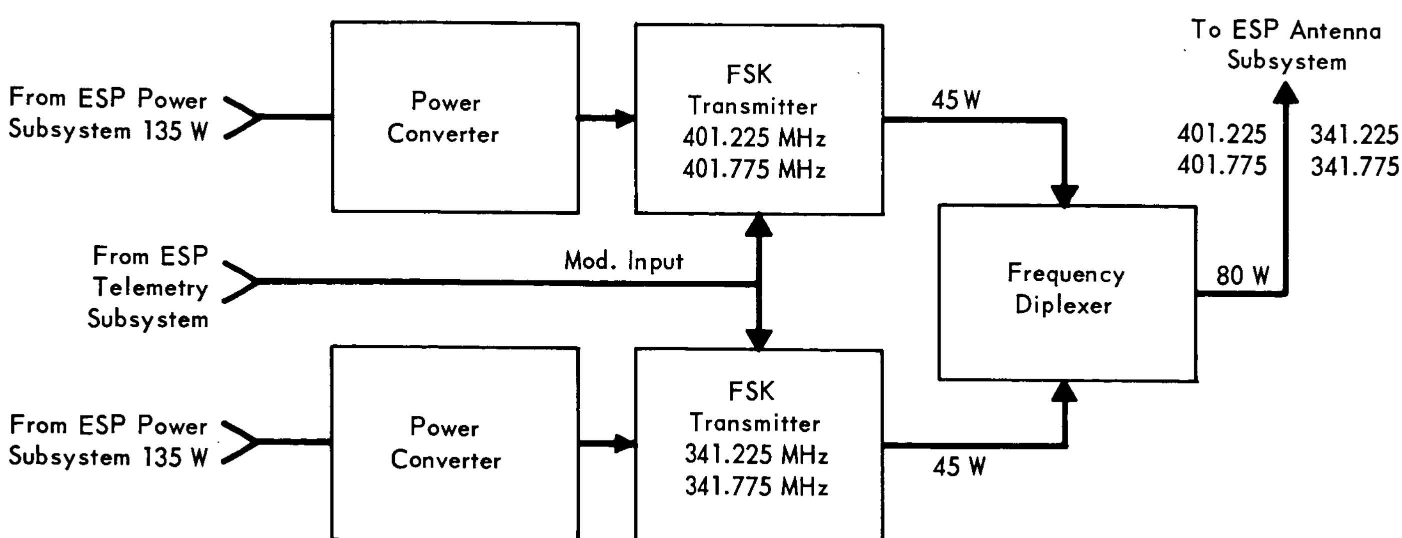


Figure 5.1-1

5.1.2 Design Requirements and Constraints - One of the most significant requirements that has imposed design constraints on the ESP Radio is the requirement for verification testing of the transmitter while enclosed within the sterilization canister. This requirement imposes the likelihood of high VSWR in RF transmission line of the ESP Radio. In order to protect the transmitter from the effects of the load mismatch, a three-port circulator is required with a suitable load connected to one of the circulator ports. This technique will effectively protect the output power amplifier stage from overload at the cost of a slight increase in line loss.

Other requirements and constraints on the design of ESP Radio have been called out in Section 4.2 of Reference 5.1.2-1.

5.1.3 Physical Characteristics - The physical characteristics of the ESP Radio are:

a.	DC Power Consumption	270 watts nominal
b.	Weight	26 pounds
c.	Volume	440 cu in
d.	Thermal Power Dissipation	190 watts

5.1.4 Operation Description - The ESP Radio will be turned on at entry and will transmit continuously until landing.

5.1.4.1 Transmitter Assemblies - The ESP Radio employs two 45 watt transmitters operating at 401.5 MHz and 341.5 MHz utilizing frequency diversity techniques to combat multipath degradation. The transmitters consist of the following functional modules:

- o Crystal Oscillators
- o FSK Modulator
- o Frequency Multiplier/AGC
- o RF Power Amplifier
- o Three-port Circulator
- o Directional Coupler and Power Detector

A block diagram of the 401.5 MHz ESP transmitter is shown in Figure 5.1-2. The block diagram of the 341.5 MHz transmitter is similar to that of the 401.5 MHz transmitter. The modules comprising the transmitters are described in the following paragraphs.

ESP TRANSMITTER BLOCK DIAGRAM

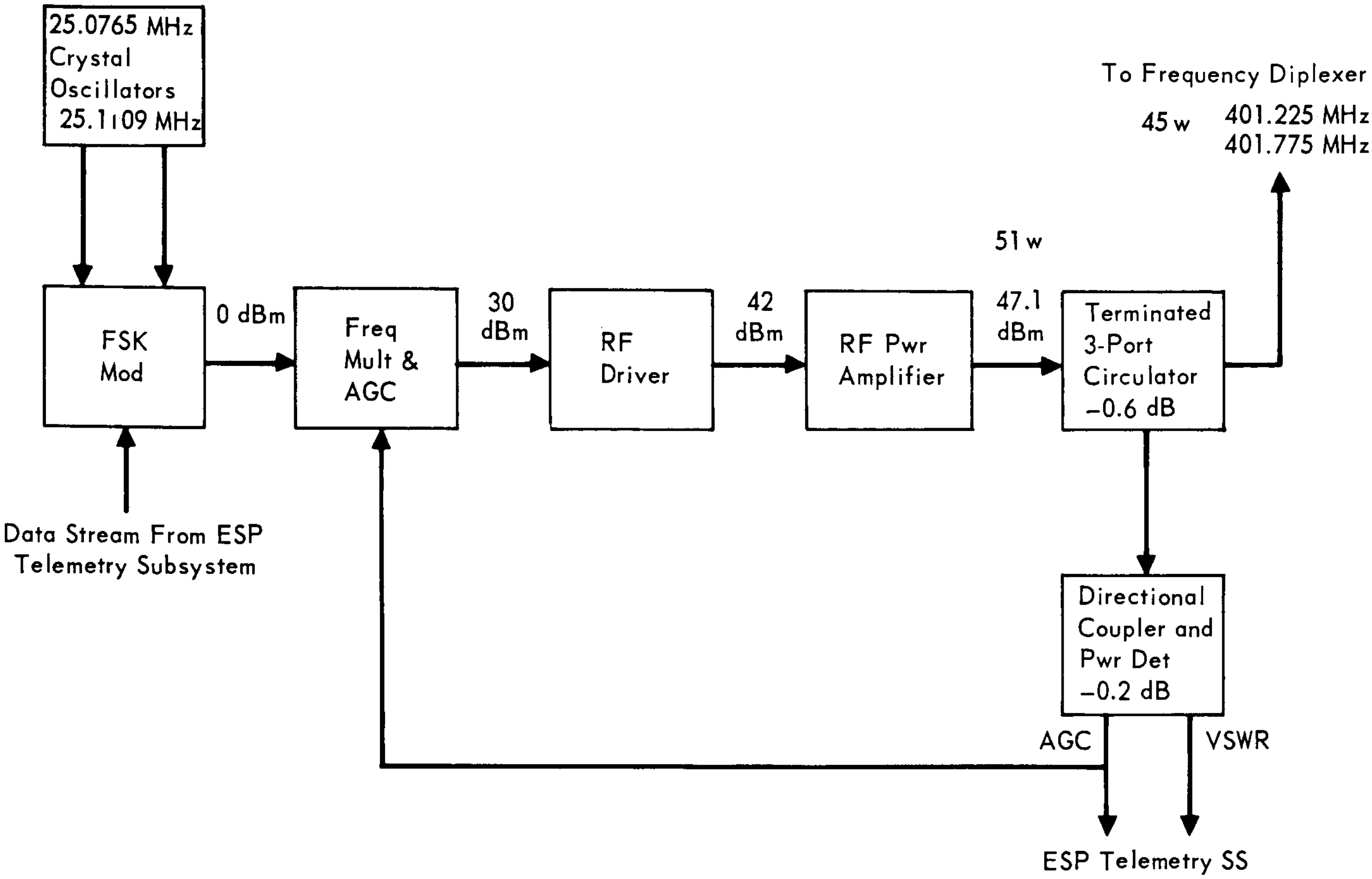


Figure 5.1-2

Crystal Oscillator - The most important design parameter of the crystal oscillator is frequency stability. The degree of frequency stability required depends upon the total frequency uncertainty tolerable in the ESP Radio Subsystem. The frequency uncertainty includes maximum Doppler shift, transmitter oscillator and spacecraft LO drifts. The maximum Doppler shift has been calculated to be approximately 6 KHz. The contribution to the total frequency uncertainty from both transmitter and receiver oscillators does not exceed 5 per cent of maximum Doppler shift. This requirement is met with temperature compensated crystal oscillators (TCXO) having a long term stability of 1×10^{-6} per year, and a short term stability of 5×10^{-9} per second. In terms of frequency stability requirements and frequency multiplier circuit complexity, the optimum choice of oscillator frequency lies in the 25 MHz region. A multiplication factor of 16 results in an output frequency of approximately 400 MHz. To obtain the FSK output frequencies of 401.225 and 401.775 and MHz, the frequencies of the crystal oscillators are 25.0765 MHz and 25.1109 MHz respectively. The 341.225 MHz and 341.775 MHz FSK frequencies are derived in the same manner, and the crystal oscillator frequencies are 21.3266 MHz and 21.3609 MHz. The advantages of TCXO's are light weight (4 ounces), low power drain (120 mW), and zero warm up time. The oscillator output level is approximately 1 milliwatt.

FSK Modulator - The FSK modulator consists of a set of RF diode switches and the associated switching control circuits, shown in Figure 5.1-3. The binary data is assumed to have a plus (Mark) and minus (space) format. The RF diode switch is essentially a ring modulator.

Frequency Multiplier - A frequency multiplication factor of 16 is obtained by cascading four transistor doubler stages. A block diagram of the frequency multiplier chain is shown in Figure 5.1-4. A power gain of approximately 6 dB per stage is obtained for an overall gain of 30 dB. A variable-gain 25 MHz amplifier stage preceding the first doubler is driven by an AGC amplifier to control the transmitter output power at a constant level.

RF Power Amplifier - The RF power amplifier stage consists of two single transistor driver stages followed by an output stage utilizing four transistors. A block diagram of the power amplifier is shown in Figure 5.1-5.

The driver stages provide a gain of 12 dB and raise the power level to 15 watts. These stages utilize medium power (2N3733) and high power (2N5017) transistors. The common emitter configuration is used for high stability and efficiency.

The output stage utilizes four RF power transistors in a parallel-hybrid configuration to provide the required 50 watts at the amplifier output.

The hybrid configuration provides isolation between amplifier branches so that in case one amplifier branch fails it will not affect the other branch. Matched pair transistors are used in each branch for optimum efficiency. With this method of combining, the amplifier imbalance due to temperature variation (-16°C to +100°C) is less than 0.1 dB as shown in Figure 5.1-6.

The RF power amplifier dissipates approximately 35 watts. Due to layout constraints the power transistors are clustered in a confined area of approximately 15 square inches. An efficient heat sink is required to keep the transistor temperature below 100°C to insure reliable operation.

Three-Port Circulator - The three-port terminated circulator is used for the protection of power transistors against high load VSWR. The UHF circulator design

BLOCK DIAGRAM FSK MODULATOR (ESP)

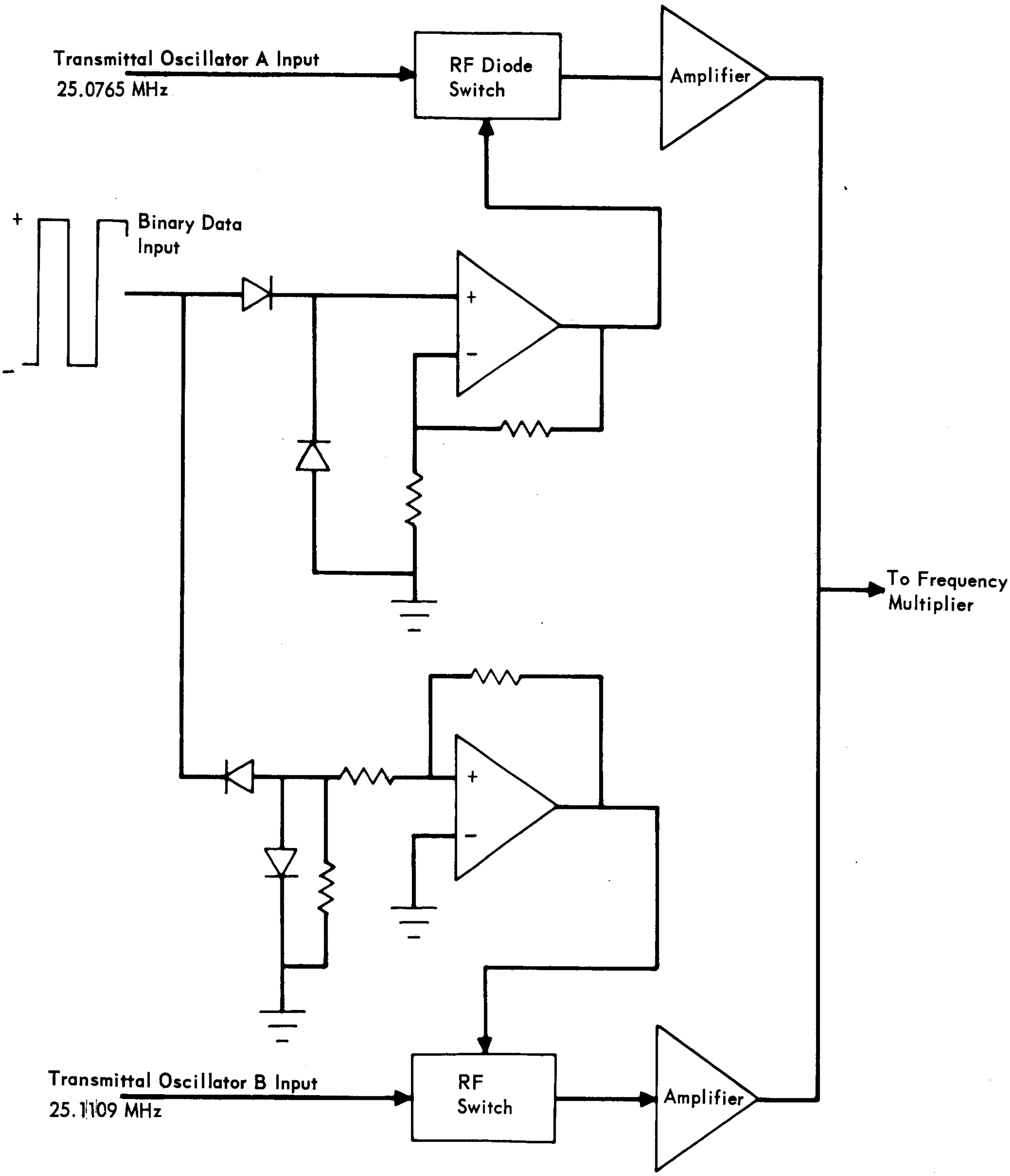


Figure 5.1-3

FREQUENCY MULTIPLIER/AGC BLOCK DIAGRAM

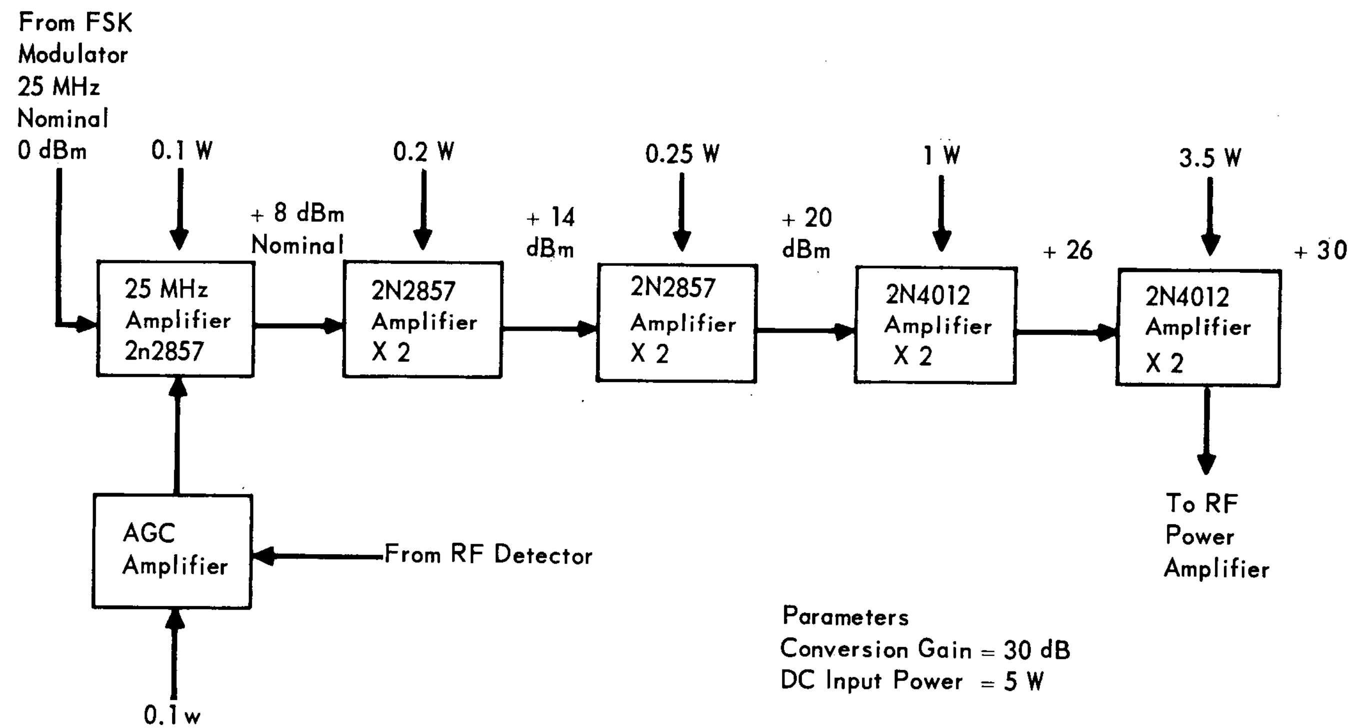


Figure 5.1-4

ESP RF POWER AMPLIFIER

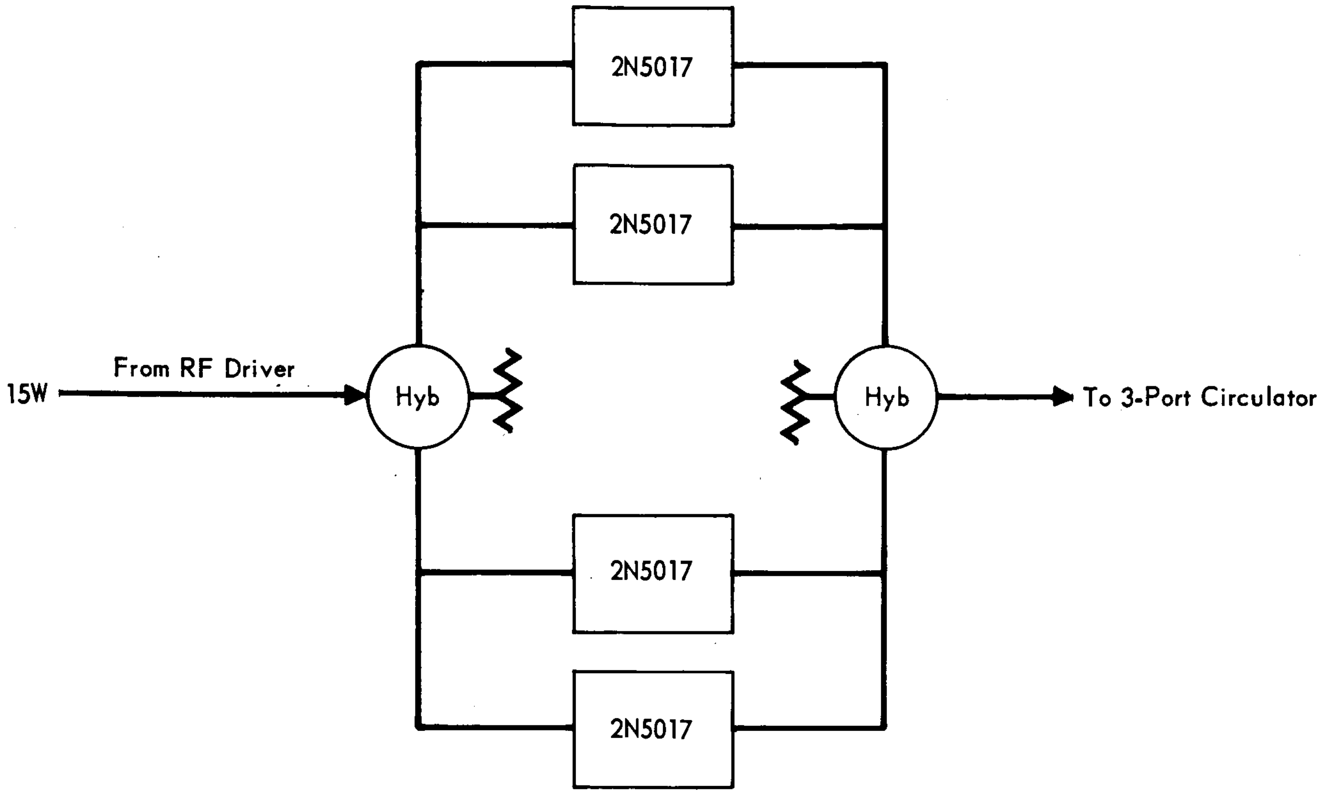


Figure 5.1–5

400 MHz SOLID-STATE POWER AMPLIFIER (3TE440) TEST DATA

TEMP °C \ POWER (watts)	INPUT (watts)	POWER LOSS DUE TO (HYBRID 1) UNBALANCE (watts)	OUTPUT (watts)	POWER LOSS DUE TO (HYBRID 2) UNBALANCE (watts)	SPECTRUM
–60	20	0.02	46	0.63	Clean
–50	20	0.02	50	0.63	Clean
–40	18	0.02	50	0.60	Clean
–16	15	0.02	50	0.50	Clean
+34	15	0.02	50	0.40	Clean
+50	15	0.02	50	0.2	Clean
+60	15	0.02	50	0.60	Clean
+100	21	0.02	50	1.00	Clean

Figure 5.1–6

employs lumped LC elements to reduce size and weight. The circulator termination is capable of dissipating 45 watts of CW power, although high VSWR is not likely to occur over a long period of time.

Directional Coupler and Power Detector - The strip-line directional coupler samples and detects the transmitter output power and provides a dc output voltage for the AGC amplifier and for telemetry purposes.

5.1.4.2 Antenna Diplexer - An antenna diplexer is used for power combining the two frequency diversity transmitters. The diplexer has the passband and impedance characteristics shown below.

f_1	400 MHz
Bandwidth	4 MHz min at 0.5 dB points
Passband ripple	0.5 dB max
Input VSWR (output terminated)	1.2 max at 400 MHz
Output VSWR (F_1 input terminated)	1.2 max at 400 MHz
Insertion loss	0.5 dB
f_2	340 MHz
Bandwidth	4 MHz min at 0.5 dB points
Passband ripple	0.5 dB max
Input VSWR (output terminated)	1.2 max at 340 MHz
Output VSWR (f_2 input terminated)	1.2 max at 340 MHz
Insertion loss	0.5 dB

5.1.4.3 Power Converter - Separate converters are used for each transmitter. Each converter is capable of delivering a minimum of 120 watts at a regulated voltage of 28 Vdc, and will operate over an input range of 23 to 40 Vdc.

The converters will utilize saturable-core transistor switching to obtain high efficiency, good regulation, and reliability.

5.1.5 Performance Characteristics - The performance characteristics of the Entry Science Package Radio are:

Transmitter #1 frequencies	401.225 MHz and 401.775 MHz
Transmitter #2 frequencies	341.225 MHz and 341.775 MHz
Frequency Stability	$\leq 1 \times 10^{-6}$ per year
	$\leq 5 \times 10^{9}$ per second
Spurious Output	$\leq$-40 dB from carrier
Power Output - Transmitter 1	40 watts nominal
Power Output - Transmitter 2	40 watts nominal

Modulated	FSK, data stream common to both transmitters
Input data format	Split phase coding
Data rate	55.86 kbps
Output power stability	$\pm$0.5 dB
DC input voltage	24 to 32 volts
DC power required	270 watts
Base plate heat sink temperatures	70°C maximum

5.1.6 Interface Definition - The ESP radio accepts split phase coded data from the telemetry subsystem at a bit rate of 55,860 bps and dc power from the power subsystem. The radio feeds two 40 watt FSK modulated carriers at 341.5 and 401.5 MHz to the antenna subsystem. The telemetry subsystem also measures engineering parameters within the radio.

5.1.7 Reliability and Safety Considerations - The following paragraphs describe the reliability and safety considerations which affect the design of the Entry Science Package radio.

5.1.7.1 Mission Success Definition - The ESP radio is required to survive pre-separation environments and successfully provide a FSK modulated, frequency diversity UHF link from the Entry Science Package to the spacecraft for transmitting engineering and television data during entry. The radio is required to operate from the beginning of the entry phase to post landing turn-off.

5.1.7.2 Reliability Model - The ESP radio has an assessed probability of successfully performing its mission of 0.9957. The functional flow model of the subsystem is shown in Figure 5.1-7, the reliability assessment matrix in Figure 5.1-8, and the mission success reliability flow model in Figure 5.1-9.

5.1.7.3 Mission Failure Modes and Effects - Figure 5.1-10 shows the fault tree for ESP radio. Both transmitters are required for full mission success to assure a reliable signal to multipath interference noise ratio. Three major subsystem failure modes are identified:

- Degraded ESP Transmission - Both transmitters operating but degraded in one or more parameters that increase the bit error rate.
- Degraded ESP to S/C Link - One of the two transmitters in a failed state with effectively no output resulting in a degraded signal to multipath interference noise ratio.
- No ESP to FSC Link - Effective loss of the ESP to FSC link. No block redundancy is utilized in this design. AGC is incorporated to provide

FUNCTIONAL FLOW DIAGRAM

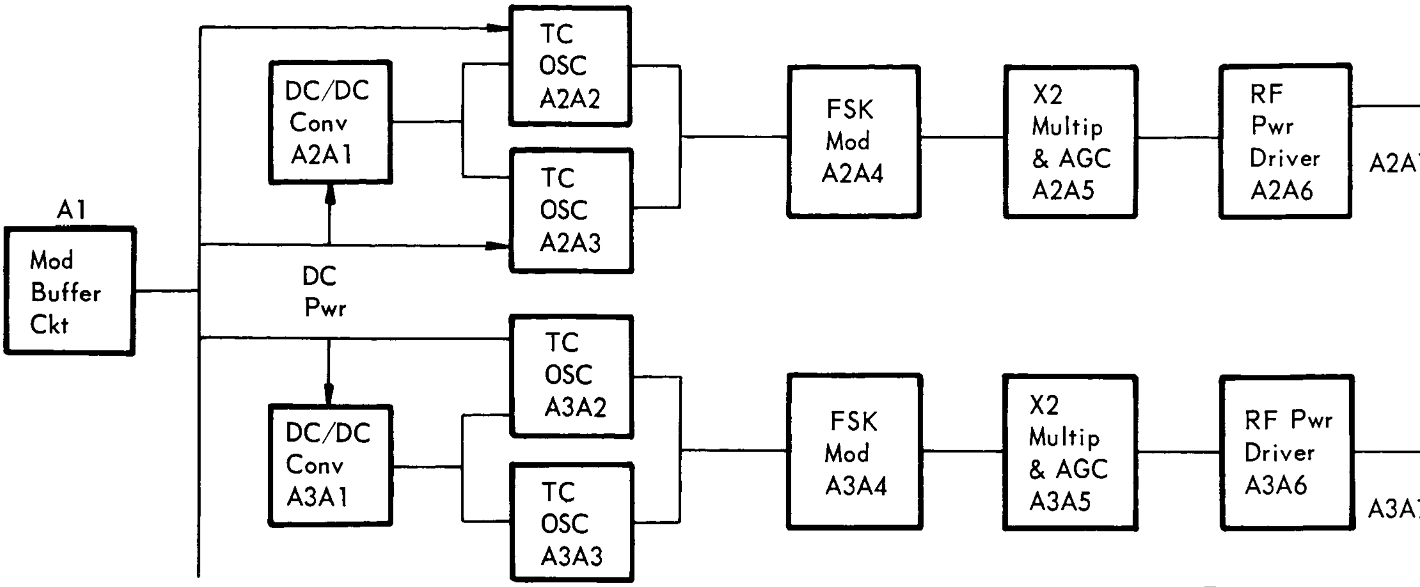
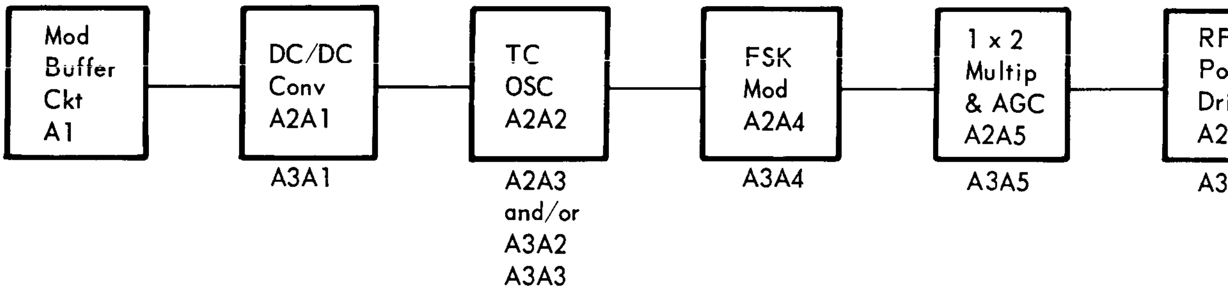


Figure 5.1–7

RELIABILITY ASSESSMENT MATRIX



Failure Rate λ	44.3	261	133.6	146.0	246.2	89.3
Σλ	88.6	261	267.2	146.0	246.2	89.3
Σ KT	115	115	115	115	115	115
Σλ·ΣKT	.0001	.00030	.000307	.000168	.000283	.000103
P_s	.9999	.9997	.999692	.999832	.999717	.999897
Total P_s	.998922 .002078					

Figure 5.1–8

RELIABILITY MODEL

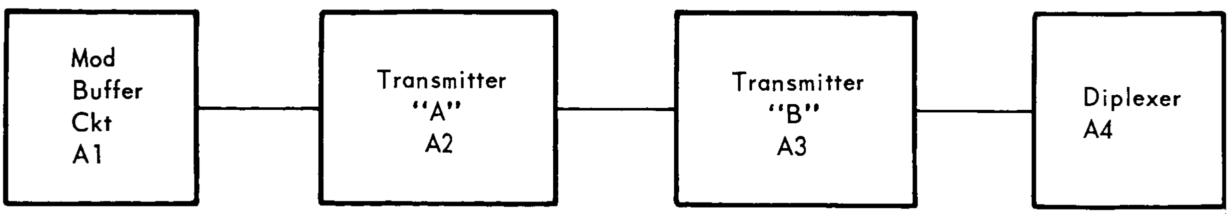
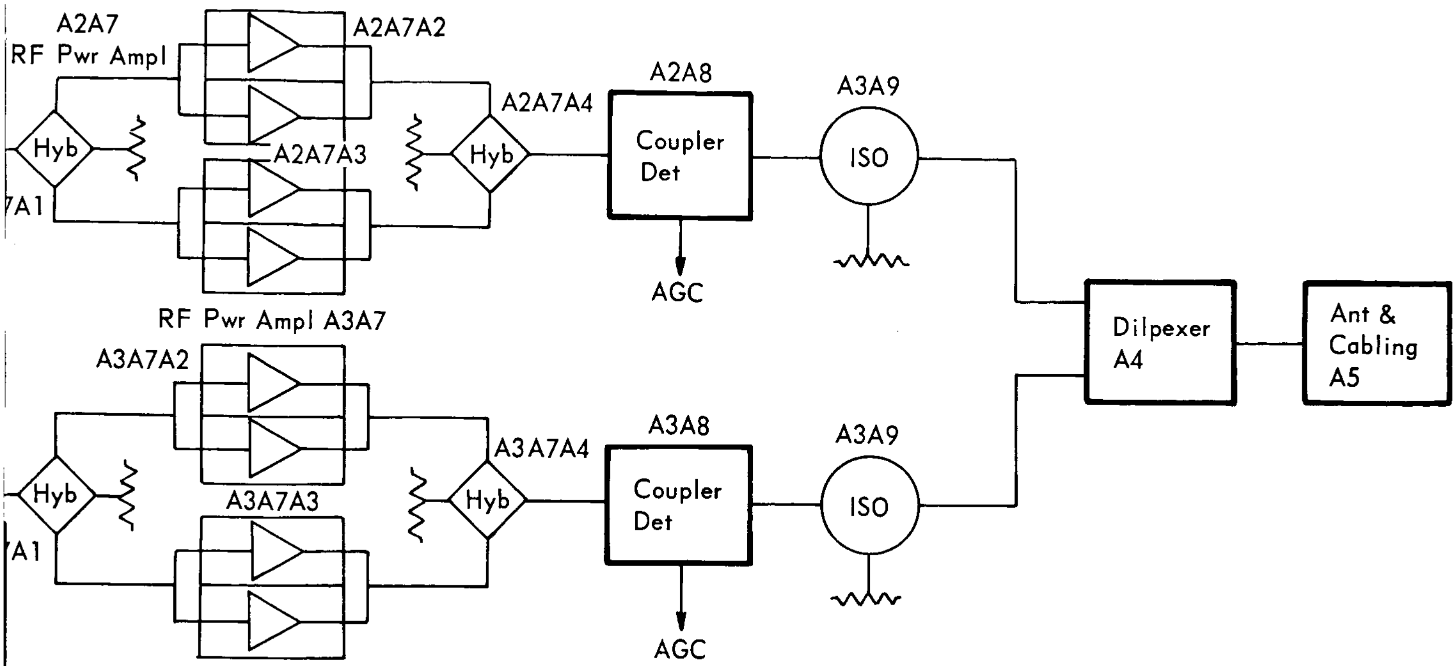
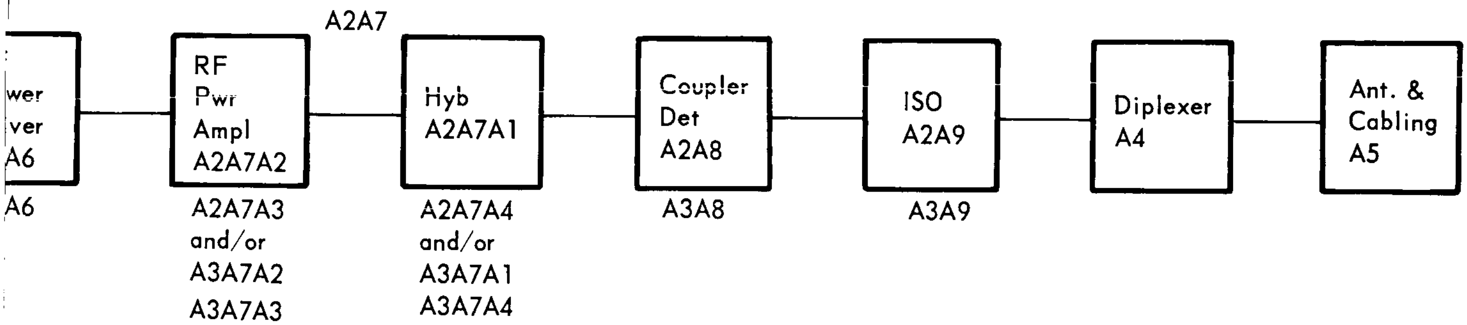


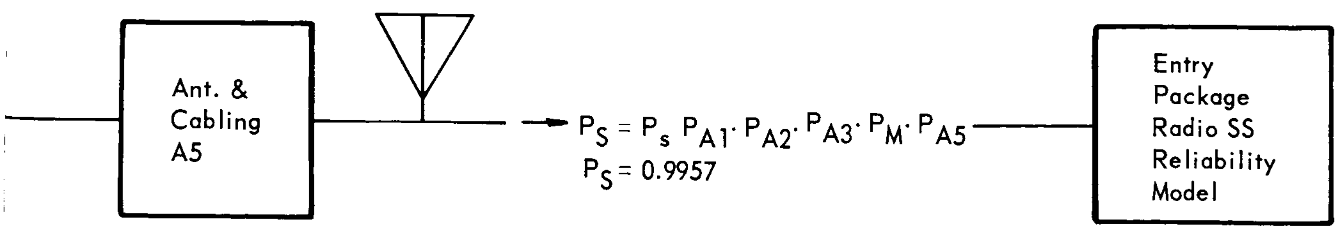
Figure 5.1–9

Figure 5.1–7, 5.1–8, 5.1–9

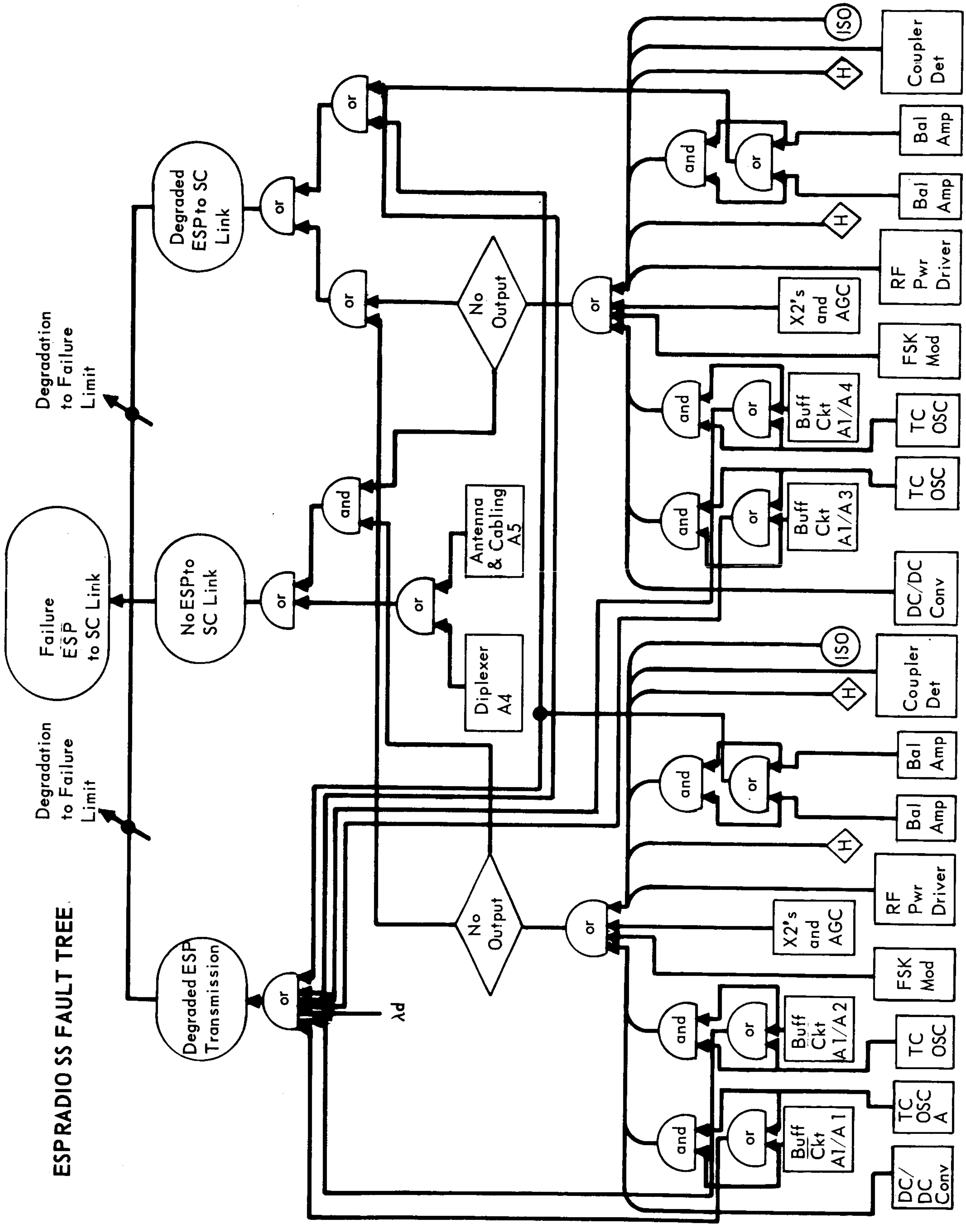




178.6	160	60.0	20	10	48.0
357.2	320	60.0	20	10	48.0
115	115	115	115	115	115
.000457	.000368	.000069	.000023	.0000115	.000055
.99954	.999632	.99993	.999977	.9999885	.99994



ESP RADIO SS FAULT TREE



Notes: λd = Performance degradation failures.

Figure 5.1-10

optimum drive levels with aging, manufacturing, and temperature-induced variances of circuit parameters.

A degree of inherent redundancy is recognized in the dual paralleled RF power amplifier mechanization. Failure of one or more of the power amplifier circuits would result in degraded power output except for failure modes that would result in severe phase imbalance. For total and reliable mission success all power amplifiers are required to function to assure an adequate signal to multipath interference noise ratio margin.

5.1.7.4 Complexity Estimate, ESP Radio - The ESP Radio is estimated to have a complexity in the order of 500 piece parts excluding hardware and the connection system parts. In Figure 5.1-11 is a table that represents a breakdown of the subsystem to generic part types.

The piece parts that are critical in assuring reliable design applications are the RF power transistors and the oscillator crystals.

5.1.7.5 Safety Considerations - No special safety considerations are noted since the voltages of the unit are in the 28 vdc region.

5.1.8 Test - Tests will be conducted as listed in the test matrix (Figure 5.1-12) in accordance with the Integrated Test Plan.

Tests will be performed prior to canister enclosure, after enclosure prior to final sterilization, after sterilization, and during pre-launch operations. All ground testing shall be performed with the use of Operational Support Equipment (OSE). In-flight checkout will be performed through the DSIF. In-flight checkout (pre-separation) is initiated by the appropriate flight capsule test programmer upon command from the DSIF and monitored through the spacecraft telemetry system.

5.1.9 Development Status - The circuit design of the various modules is conservative and utilizes proven design techniques. No long lead time development will be required at the module level. Qualification of parts and suppliers will require tests which include sterilization and decontamination environments in addition to requirements for parameter stability, long life, and failure rate.

5.2 SPACECRAFT-MOUNTED ESP SUPPORT RADIO

5.2.1 Equipment Identification and Usage - The spacecraft-Mounted (FSC-MTD) ESP support radio is a part of the UHF telecommunications relay link. It is used in the Flight Spacecraft (FSC) for receiving the Flight Capsule (FC) television and engineering data for retransmission to Earth via the Spacecraft Radio Subsystem. The spacecraft mounted ESP support radio consists of the following subassemblies:

ENTRY PACKAGE RADIO COMPLEXITY ESTIMATE

GENERIC PART TYPE	λ	η	$\eta\lambda$
1.0 Capacitors			
Ceramic/Glass	1.0	46	46.0
Tantalum Solid	2.0	3	6.0
Variable Piston	10.0	21	210.0
Feed Thru	5.0	7	35.0
2.0 Diodes			
General Purpose/Load Power	1.1	25	27.5
Zener/Reference	10.0	3	30.0
3.0 Resistors			
Metal Film	0.3	15	4.5
Carbon Comp	0.1	45	4.5
4.0 Transistors			
Small Sig/Med Pwr/Swt	5.0	11	55.0
Power	50.0	3	150.0
Power/RF	50.0	5	250.0
5.0 Chokes and Coils			
RF Chokes	7.5	23	172.5
LV Coil	10.0	19	190.0
6.0 Transformers			
RF/IF/Signal	5.0	1	5.0
Low Voltage/Power	5.0	5	25.0
7.0 Crystals	65.0	2	130
8.0 Filters	5.0	10	50.0
9.0 IC–Linear/Op Amp	30.0	1	30.0
10.0 Probe RF	.4	1	0.4
11.0 Band Pass Filter	60.0	1	60.0
Parts Subtotal			
One Transmitter		247	1481.4
Both Transmitters		494	2962.8
RF Components			
Coupler	60.0	2	120.0
Isolator	20.0	2	40.0
Diplexer	10.0	1	10.0
Antenna and Cabling	48.0	1	48.0
RF Component Subtotal		6	218.0
Total		500	3180.8

Figure 5.1–11

TEST REQUIREMENTS MATRIX – ENTRY SCIENCE PACKAGE

RADIO	Non-Operative Test	RF Power Output	Power Amplifier Collector Current	Power Amplifier Collector Current	Power Amplifier Collector Current	Power Amplifier Collector Current	Power Amplifier Collector Current	Power Supply Voltage	Case Temp A	Case Temp B	RF Drive (Detected)	DC Current Drain	Modulation Input	Transmitter No. 1 Frequency	AGC Voltage	Spurious Output	Detected RF Power Output	RF Power Stability	Frequency Stability	Transmitter No. 2 Frequency	VSWR Monitor
ACCY REQ.		0.2 dB	(a) 2%	(b) 2%	(c) 2%	(d) 2%	(e) 2%	1%	2%	2%	2%	5%	5%	1x10−8	2%	1 dB	0.5 dB	0.2 dB	1x10−8	1 x 10−8	2%
Telemetry			X	X	X	X	X	X	X	X	X	X			X		X				X
System Test (Pre-Canister)		X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X	X
System Test (With Canister)			X	X	X	X	X	X	X	X	X	X		X	X		X		X	X	X
Pre-Launch			X	X	X	X	X	X	X	X	X	X		X	X		X			X	X
In-Flight Checkout			X	X	X	X	X	X	X	X	X	X			X		X				X

Figure 5.1–12

a. Frequency Diplexer
b. 401.5 MHz Receiver
c. 341.5 MHz Receiver
d. Diversity Combiner
e. Bit Synchronizer
f. Power Converter

A block diagram of the FSC-MTD ESP support radio is shown in Figure 5.2-1. The receivers accept the 401.5 MHz and 341.5 MHz frequency diversity ESP radio signals from the spacecraft ESP antenna. The received signals are translated down to intermediate frequencies for amplification and demodulation. The demodulated FSK signals are fed to the diversity combiner. The video output from the diversity combiner is fed to the bit synchronizer. The bit synchronizer processes the video signal and presents a serial non-return to zero (NRZ) bit stream along with synchronization pulses to the Capsule Bus Data Distribution Unit. The power converters accept the 23 to 40 vdc input from the Spacecraft Power Subsystem and convert the power into separate isolated -15 vdc outputs to each receiver, diversity combiner, and bit synchronizer.

5.2.2 Design Requirements and Constraints - The FSC-MTD ESP Radio must share the exclusive frequency band in the 400 MHz region with the Capsule Bus Radio Subsystem. This necessitates the use of a filter to attenuate the CB radio signals.

Because of the multipath environment, AGC design considerations must include fading. The gain of the two frequency diversity receivers must be nearly the same throughout the dynamic range. This requirement is met by using identical IF amplifiers and AGC circuits for both receivers.

5.2.3 Physical Characteristics - The physical characteristics are shown in Figure 5.2-2.

5.2.4 Operation Description - The FSC-MTD ESP Support Radio will be turned on prior to FSC-FC separation for in-flight checkout and will remain on from separation until landing. It will not receive an RF input until the ESP Radio is turned on at entry.

Sub-Assembly Description

Frequency Diplexer - The diplexer separates the 341.5 MHz and 401.5 MHz frequency diversity signals from the FSC/ESP antenna and feeds them into separate receivers. The diplexer characteristics are shown in Figure 5.2-3.

Receivers - Except for the RF front end the 341.5 MHz and 401.5 MHz receiver

SPACECRAFT-MOUNTED ESP SUPPORT

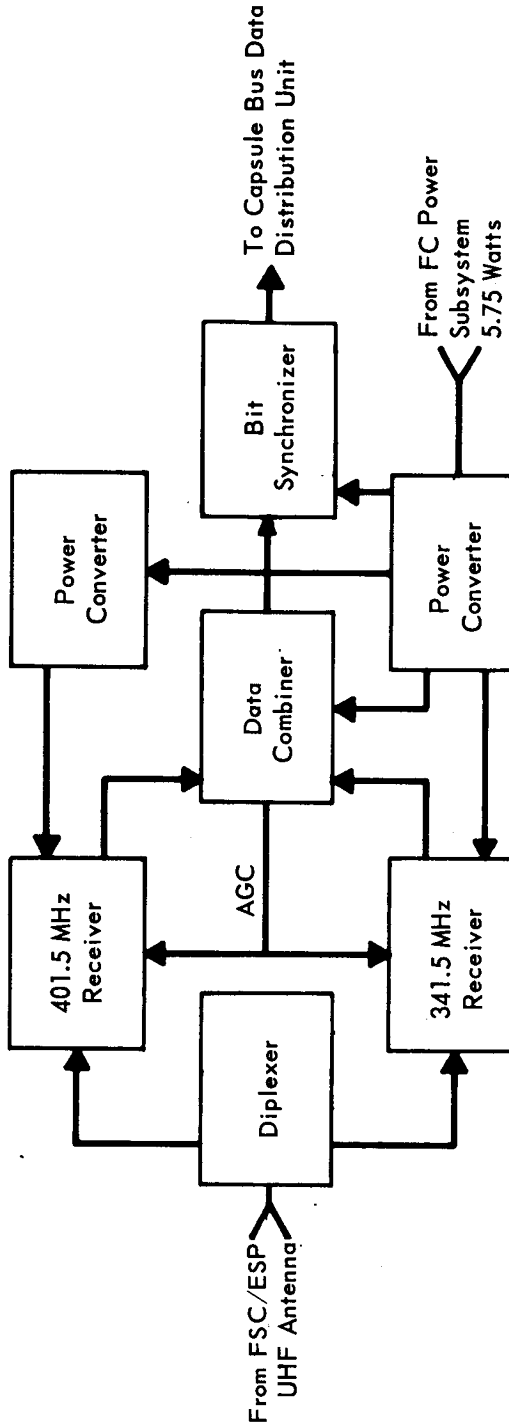


Figure 5.2-1

SPACECRAFT-MOUNTED ESP SUPPORT RADIO PHYSICAL CHARACTERISTICS

ASSEMBLY	401.5 MHz RECEIVER	341.5 MHz RECEIVER	DIVERSITY COMBINER	BIT SYNCHRONIZER	POWER CONVERTER	FREQUENCY DIPLEXER	HOUSING	TOTAL
Size (inches)	5 x 8 x 1.5 (60 in.^{3})	5 x 8 x 1.5 (60 in.^{3})	2 x 2.5 x 1.5 (7.5 in.^{3})	2 x 4 x 1.5 (12 in.^{3})	2 x 4 x 1.5 (12 in.^{3})	2 x 4 x 1 (8 in.^{3})	10 x 8 x 2.5 (200 in.^{3})	10 x 8 x 2.5 (200 in.^{3})
Weight (pounds)	3.6	3.6	0.45	0.72	1.0	0.8	1.8	10.97
Input Power (Watts)	2	2	0.30	0.30	5.75	–	–	5.75
Power Dissipation (Watts)	2	2	0.30	0.30	1.15	–	–	5.75

Figure 5.2-2

ESP RADIO DIPLEXER CHARACTERISTICS

Input frequencies	341.5 MHz and 401.5 MHz nominal
Imput impedance	50 ohm nominal
Input VSWR	
at 341.5 MHz	1.2 max (341. 5MHz output terminated)
at 401.5 MHz	1.2 max (401.5 MHz output terminated)
Output VSWR	
at 341.5 MHz	1.2 max (Input terminated)
at 401.5 MHz	1.2 max (Input terminated)
Bandpass at 341.5 MHz output:	
Bandwidth	4 MHz min at – 1.0 dB points 60 MHz max at – 60 dB points
401.5 MHz rejection	60 dB min
Bandpass at 401.5 MHz Output:	
Bandwidth	4 MHz min at – 1.0 dB points 60 MHz max at – 60 dB points
341.5 MHz rejection	60 dB min
Insertion Loss	0.5 dB maximum.

Figure 5.2-3

components are identical. Each receiver consists of the following modules:

a. RF Front End

b. First IF

c. Second IF

d. Demodulator

e. Local Oscillator

A block diagram of the 401.5 MHz receiver is shown in Figure 5.2-4. The threshold signal for the receiver is -110.2 dBm. The open loop gain between the square law detector input and the receiver input is 100 dB. The nominal power level at the square law detector is -10 dBm. The receiver gain distribution is shown in Figure 5.2-5. The approximate receiver transfer characteristic is shown in Figure 5.2-6. The detector output is held constant within +2 dB by the AGC.

RF Front End - The RF front end includes a preselector, RF amplifier, first mixer, and filter. The RF amplifier gain is 25 dB and the noise figure is 3 dB at 400 MHz. The overall receiver noise figure is less than 4 dB. The mixer loss is less than 7 dB when operated with a LO drive of 0.5 milliwatt at 371.5 MHz.

First IF - The first IF module includes a 30 MHz IF amplifier and the second mixer. The 30 MHz IF gain is 53 dB maximum and -48 dB minimum, which is controlled by the AGC. The basic design is similar to the Mariner 69 Spacecraft Radio receiver first IF. An additional 30 MHz output is provided for noise figure measurement. This output is terminated with a shielded load when not making NF measurement.

Second IF - The second IF module includes the 6.78 MHz amplifier, and the distribution amplifier. The IF bandwidth is 1 MHz. IF gain including the distribution amplifier is 42 dB.

Demodulator - The FSK demodulation is implemented by a set of predetection filters and square law detectors.

The predetection filter bandwidth is 120 kHz.

The minimum frequency separation for the FSK carriers is determined by the predetection bandwidth, filter selectivity ratio and the signal spectrum width. In order to avoid crosstalk, the FSK carrier separation for the ESP Radio Subsystem is approximately five times the predetection bandwidth, or 550 kHz.

Local Oscillator - The local oscillator generates coherently a 23.1375 MHz and a 370.2 MHz signal for the second and first mixers, respectively. The 370.2 MHz signal is derived from the 23.1375 MHz crystal oscillator by multiplication. The crystal oscillator is temperature-compensated, with stabilities of 1×10^{-6} per

401.5 MHz RECEIVER BLOCK DIAGRAM

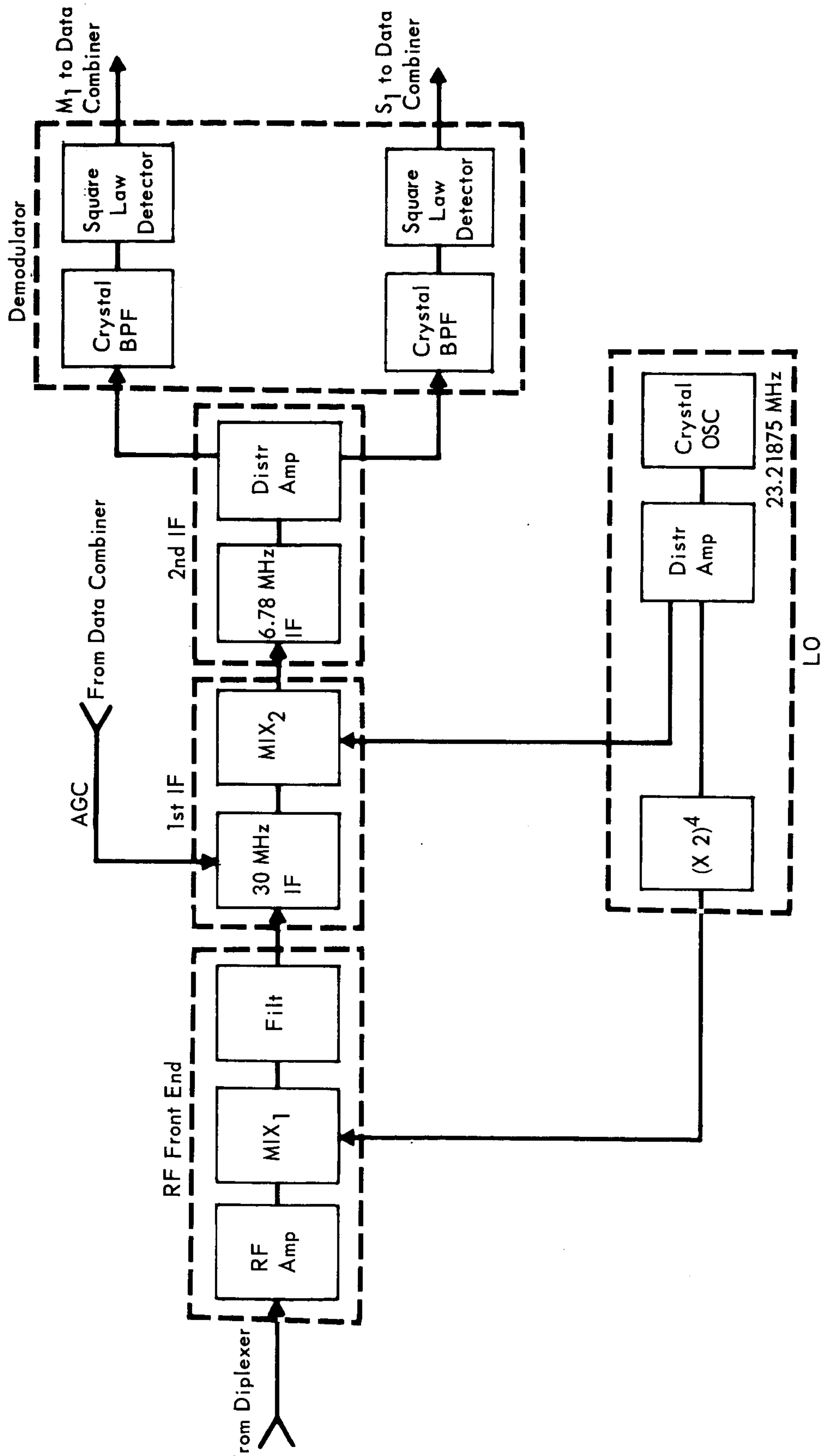


Figure 5.2-4

PARAMETERS	POWER GAIN (dB)
RF Amplifier	25
1st Mix	-7
BPF (Crystal)	-6
1st IF and 2nd Mix	-48 to + 53
2nd IF	42
Predetection Filter	-6
	101 dB (Min Input Signal)
	0 dB (Max Input Signal)

Figure 5.2-5

DYNAMIC CHARACTERISTICS

INPUT LEVEL (dBm)	CHARACTERISTICS	OUTPUT LEVEL (dBm)			
		RF 400 MHz or 340 MHz BW = 2 MHz No = −170 dBm (Include NF)	1st IF 30 MHz BW = 2 MHz NF = 3 dB G = −48 to +53 dBm	2nd IF 6.78 MHz BW = 1 MHz G = 42 dB NF = 3 dB No = −69 dBm	DET. INPUT 6.5 MHz and 7.056 MHz BW = 120 KHz
−111	Signal Noise	−99 −95	−46 −42	−4 −3	−10 −18.2
−101	Signal Noise	−89 −95	−46 −52	−4 −13	−10 −28.2
−81	Signal Noise	−69 −95	−46 −72	−4 −33	−10 −48.2
−61	Signal Noise	−49 −95	−46 −92	−4 −53	−10 −68.2
−41	Signal Noise	−29 −95	−46 −	−4 −69	−10 −84.2
−21	Signal Noise	−9 −95	−46 −	−4 −69	−10 −84.2

Figure 5.2-6

year and 5 x 10^{-9} per day.

Diversity Combiner - The demodulated data stream from the receiver square law detectors are added by the diversity combiner as shown in Figure 5.2-7. One operational amplifier provides the necessary amplification and differential output of the detected FSK, split phase coded binary data. The other operational amplifiers are used for amplification, integration and isolation of the AGC signal for the receivers. With a time constant of 0.2 milliseconds the AGC would keep a constant IF output at the envelope detector even when fast fading occurs.

Power Converter - The power converter accepts the dc input from the Spacecraft Power Subsystem and converts the power into separate isolated -15 vdc outputs to each receiver, and video circuitry (diversity combiner and bit synchronizer). The converters utilize saturable core transistor switching techniques exhibiting a high degree of performance.

Bit Synchronizer - The bit synchronizer demodulates the video output from the diversity combiner and presents a serial non-return to zero (NRZ) bit stream along with synchronization pulses to the Capsule Bus Data Distribution Unit. The following circuits make up the bit synchronizer.

a. Input Circuits
b. Data Channel
c. Time Gate
d. Phase Locked Loop

A block diagram of the bit synchronizer is shown in Figure 5.2-8.

The input circuits accept the diversity combiner output and filter the waveform with a low pass filter which cuts off at a frequency equal to twice the data rate (111 KHz). Automatic gain control and dc cancellation are also incorporated before the signal is fed to the data channel. This signal is limited, differentiated and rectified (producing pulses at the zero crossings of the input) and fed to the time gate.

The time gate selects those pulses corresponding to the transition in the middle of each split phase bit by responding to those pulses which recur at the same position in each bit period and gates these pulses into the phase locked loop.

The phase locked loop integrates the jitter out of the gated input pulses and creates a square wave which is coherent with the bit period. This square wave is fed to the data channel.

DIVERSITY COMBINER

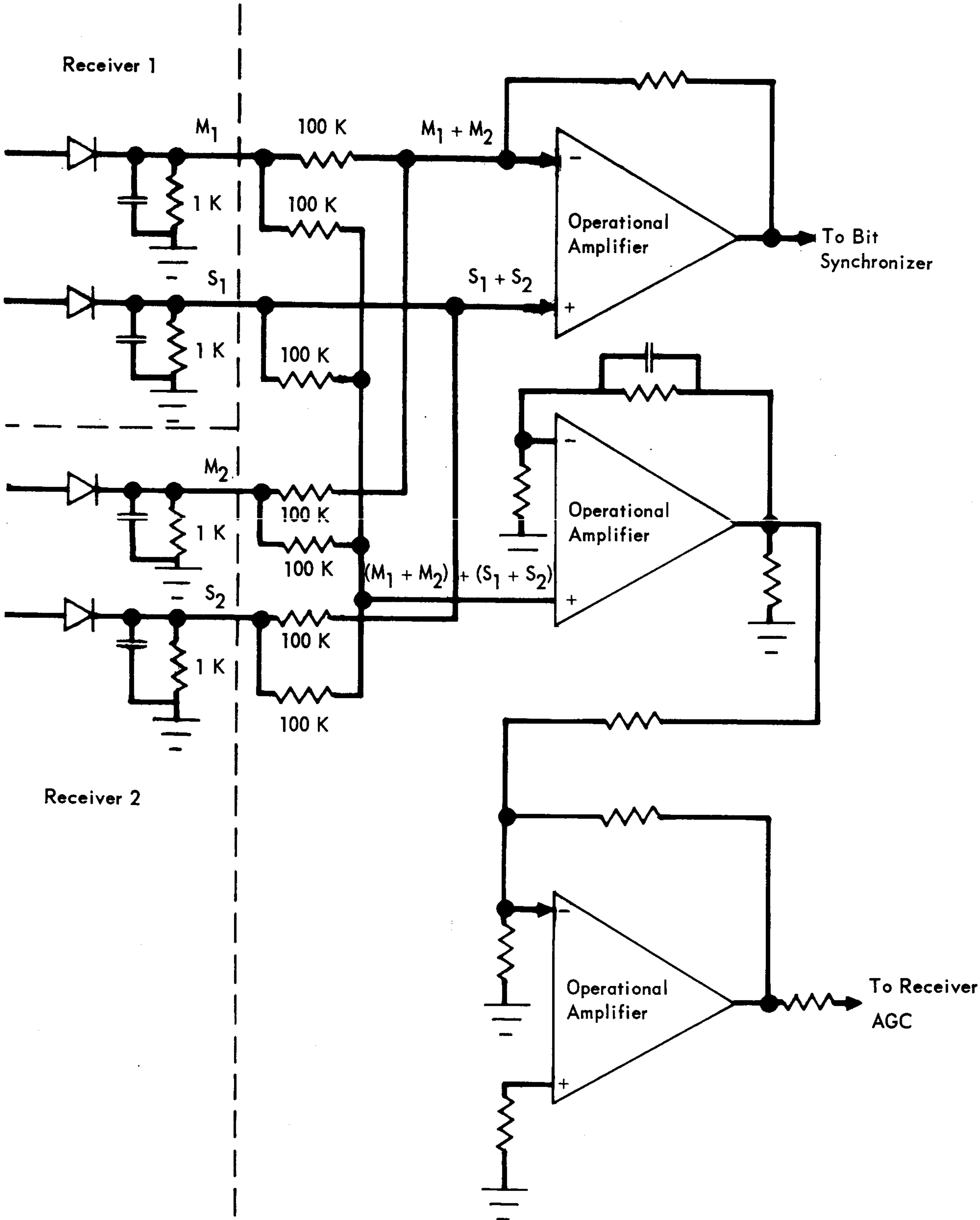
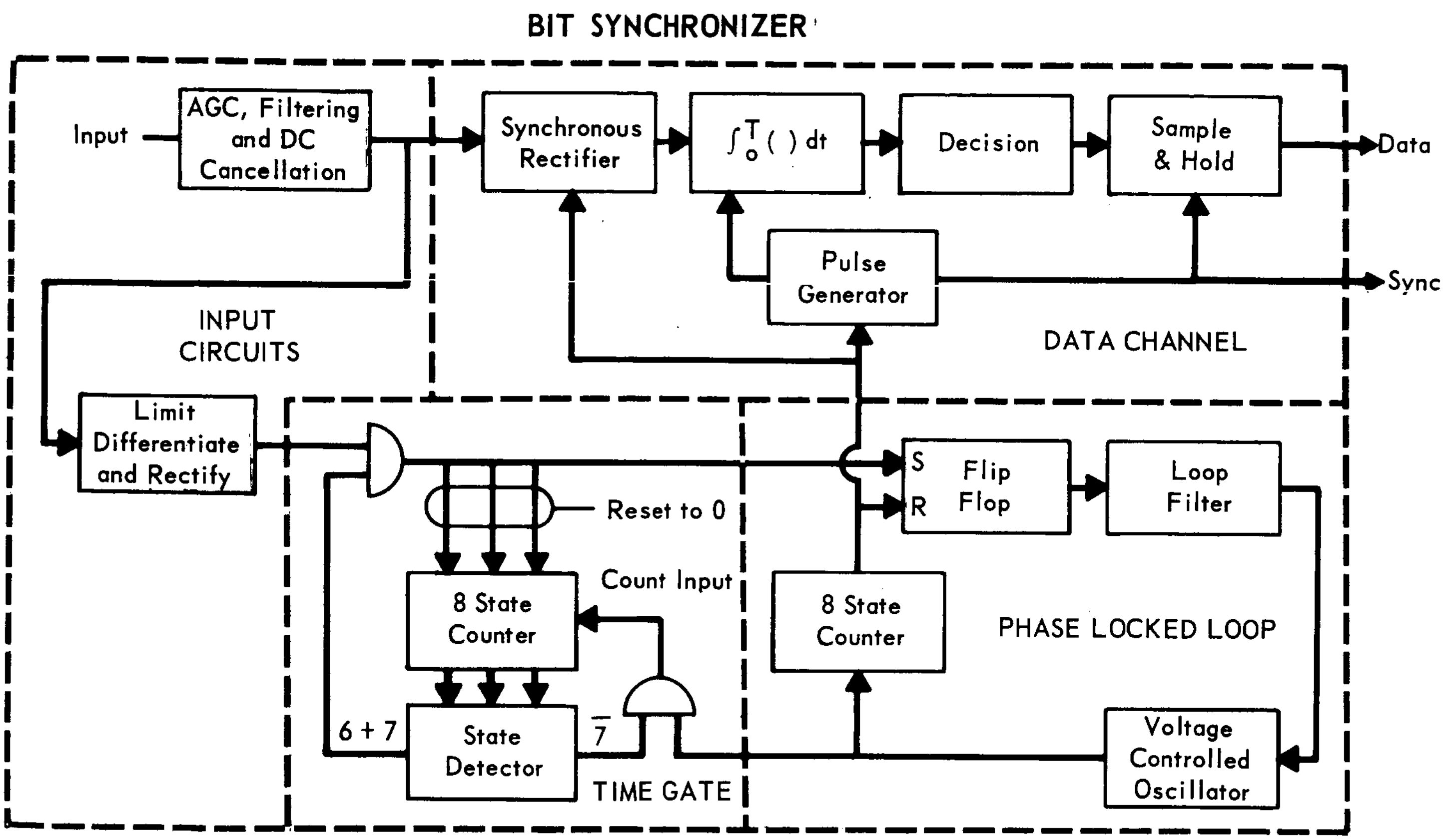


Figure 5.2-7



BIT SYNCHRONIZER
Input
AGC, Filtering and DC Cancellation
Synchronous Rectifier
$\int_0^T (\ ) dt$
Decision
Sample & Hold
Data
Pulse Generator
Sync
INPUT CIRCUITS
DATA CHANNEL
Limit Differentiate and Rectify
Reset to 0
S
R
Flip Flop
Loop Filter
Count Input
8 State Counter
8 State Counter
PHASE LOCKED LOOP
6 + 7
State Detector
$\overline{7}$
TIME GATE
Voltage Controlled Oscillator

Figure 5.2-8

The data channel converts the input split phase data into non-return to zero (NRZ) data and integrates this NRZ data stream over each bit period using the square wave reference from the phase locked loop for these operations. The reconstructed bit stream and timing pulses are fed to the Capsule Bus Data Distribution Unit.

The parameters which affect the performance of the bit synchronizer are the ratio of nominal time gate opening to bit duration, the ratio of loop noise bandwidth to bit rate and the damping constant of the loop. The nominal time gate opening has been taken to be the rise time of the filtered data waveform in order to ensure passage of the desired pulses which have been shifted in time by the noise but exclude the extraneous pulses produced by noise which is much greater than the signal.

The loop noise bandwidth has been set at the sum of the frequency uncertainties of the received data rate and the VCO in order to provide fast acquisition and yet minimize the sensitivity to jitter of the time gate as it is perturbed by noise.

In accordance with standard tracking loop design the loop damping factor has been set at 0.707.

5.2.5 Performance Characteristics - The performance characteristics of the FSC-MTD ESP Support Radio are listed in Figure 5.2-9.

5.2.6 Interface Definition - The interface diagram for the FSC-MTD ESP Support Radio is shown in Figure 5.2-10.

5.2.7 Reliability and Safety Considerations - The following paragraphs describe the reliability and safety considerations of the FSC-MTD ESP Support Radio.

5.2.7.1 Mission Success Definition - The FSC-MTD ESP Radio mission is to provide reception, detection and bit sychronization capability for a UHF FSK modulated frequency diversity link from the Entry Package to the spacecraft for receiving the Flight Capsule (FC) television and engineering data.

5.2.7.2 Reliability Model - The FSC-MTD ESP Support Radio has an assessed reliability of a 0.9973 probability of successfully performing its required mission. The functional flow model is shown in Figure 5.2-11, the functional reliability assessment matrix is shown in Figure 5.2-12, and the mission success flow model is shown in Figure 5.2-13.

5.2.7.3 Mission Failure Modes and Effects - In Figure 5.2-14 is shown the mission fault tree for the ESP Radio. Both receivers are required for full mission success to assure a reliable signal to multipath interference noise ratio. Three major failure modes are identified:

PERFORMANCE CHARACTERISTICS - SPACECRAFT MOUNTED ESP SUPPORT RADIO

RF Frequencies	
a. Receiver No. 1	401.5 MHz nominal
b. Receiver No. 2	341.5 MHz nominal
Noise Figure	≤4.0 dB
Threshold Signal	-109.7
Dynamic Range	90 dB
Maximum Input Level	+20 dBm
Image Rejection	40 dB min.
Predetection Bandwidth	120 kHz
Post Detection Bandwidth	120 kHz
1st IF Frequency	30 MHz nominal
1st IF Bandwidth	2 MHz
2nd IF Frequency	6.8 MHz nominal
2nd IF Bandwidth	1 MHz
Predetection Filter Frequencies	6.5 MHz and 7.056 MHz
Local Oscillator Frequencies	
a. Receiver No. 1	375.1 MHz (1st mix) 23.21875 MHz (2nd mix)
b. Receiver No. 2	371.5 MHz (1st mix) 23.21875 MHz (2nd mix)
AGC Time Constant	0.2 millisecond
DC Input Voltage	-15 Volts Regulated
DC Input Power	6 watts maximum
Output Data Format	NRZ bit stream and synchronization pulses

Figure 5.2-9

S/C-MTC ESP SUPPORT RADIO INTERFACE DIAGRAM

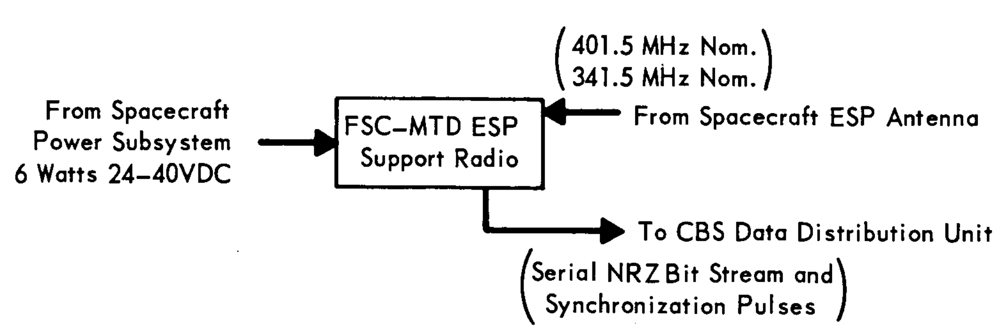
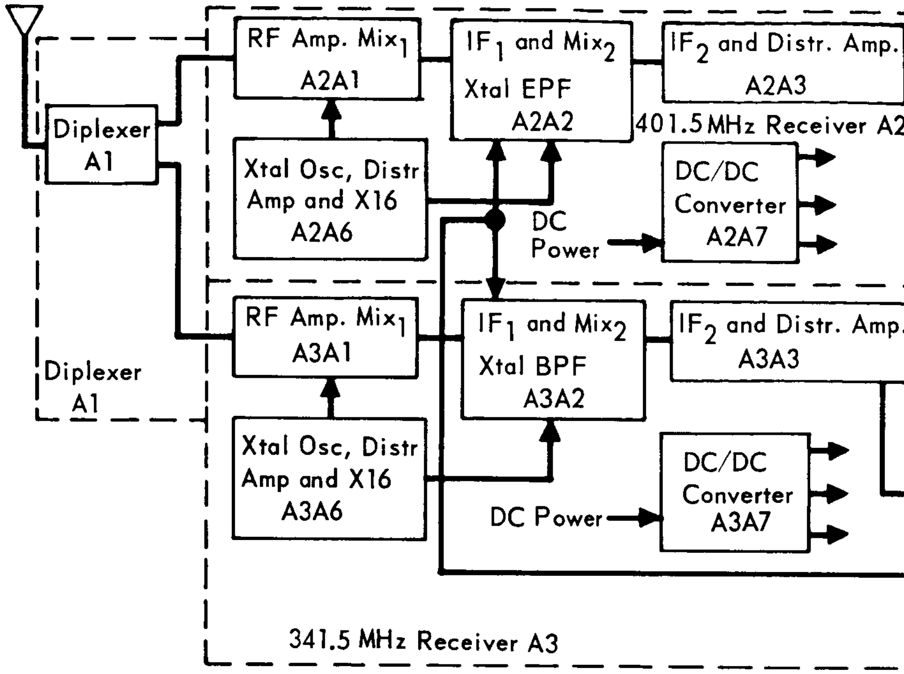


Figure 5.2-10

RADIO SUBSYSTEM

A. FUNCTIONAL FLOW



B. FUNCTIONAL RELIABILITY ASSESSMENT MATRIX

	Diplexer A1	RF Amp + A2A1 A3A1	IF$_1$ + A2A2 A3A2	IF$_2$ + A2A3 A3A3	Xstal BPF and Square Law Det. A2A4 A2A5 A3A4 A3A5	X C
λ	10	201	369	307	352	
ϵKt	115	115	115	115	115	
$\lambda\epsilon Kt$	.0000115	.000221	.000405	.000337	.000774	
P_s	$.9_49885$	$.9_39779$	$.9_39595$	$.9_39663$	$.9_39226$	

C. FUNCTIONAL RELIABILITY MODEL

Diplexer A1	400 MHz Receiver A2	Buffer Ckt. A4A1A2 A4A1A2	340 MHz Receiver	Buffer Ckt. A4A1A3 A4A1A4	Summer Amp. A4A1A5
.0000115	.00223	$.0_50690$	.00223	$.0_50690$	.0000341
$.9_49885$	.99777	$.9_59310$	.99777	$.9_59310$	$.9_49659$
				(X2)	

$\Sigma(\lambda)(\Sigma Kt) = .00275$

5-27-1

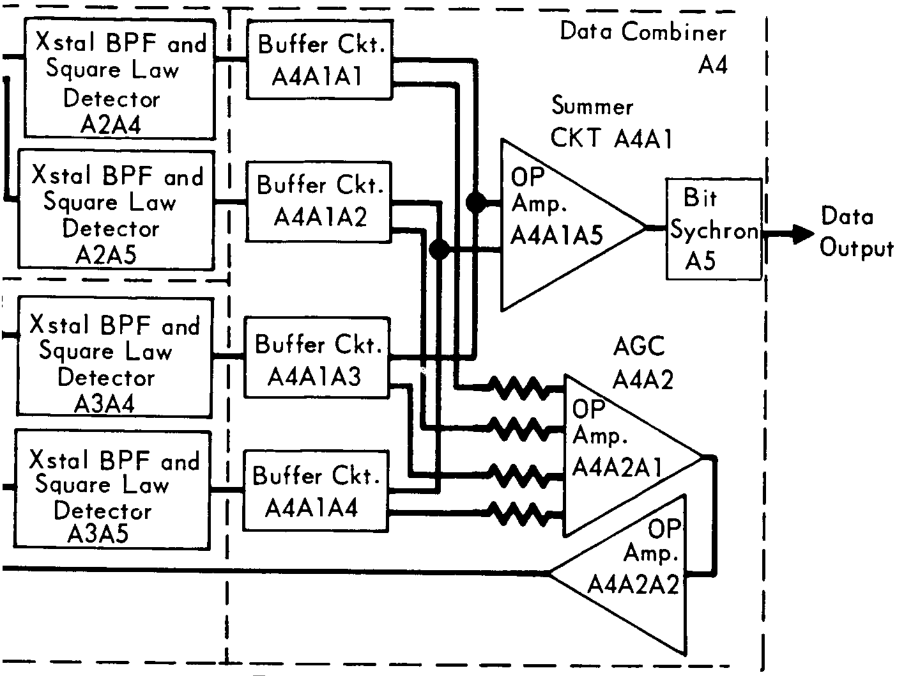


Figure 5.2–11

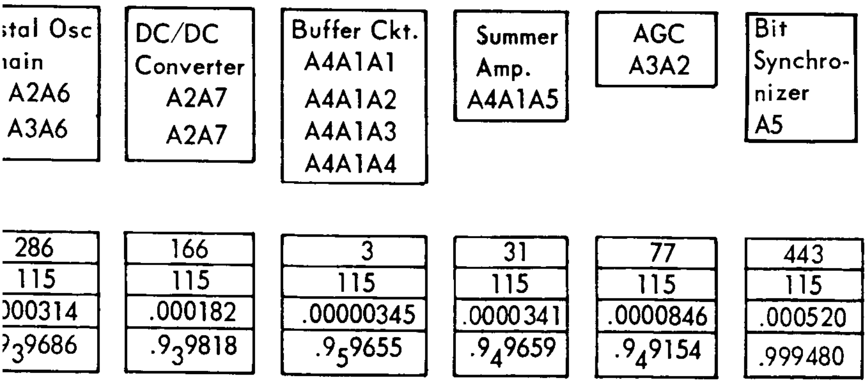
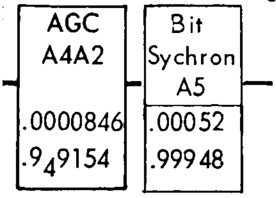


Figure 5.2–12



$$P_s = P_{A1}, P_{A2} \cdot P_{\substack{A4A1A1 \\ A4A1A2}} \cdot P_{A3} \cdot P_{\substack{A4A1A3 \\ A4A1A4}} \cdot P_{A4A1A5}\, P_{A4A2}$$

$$P_s = 0.99725$$

Figure 5.2–13

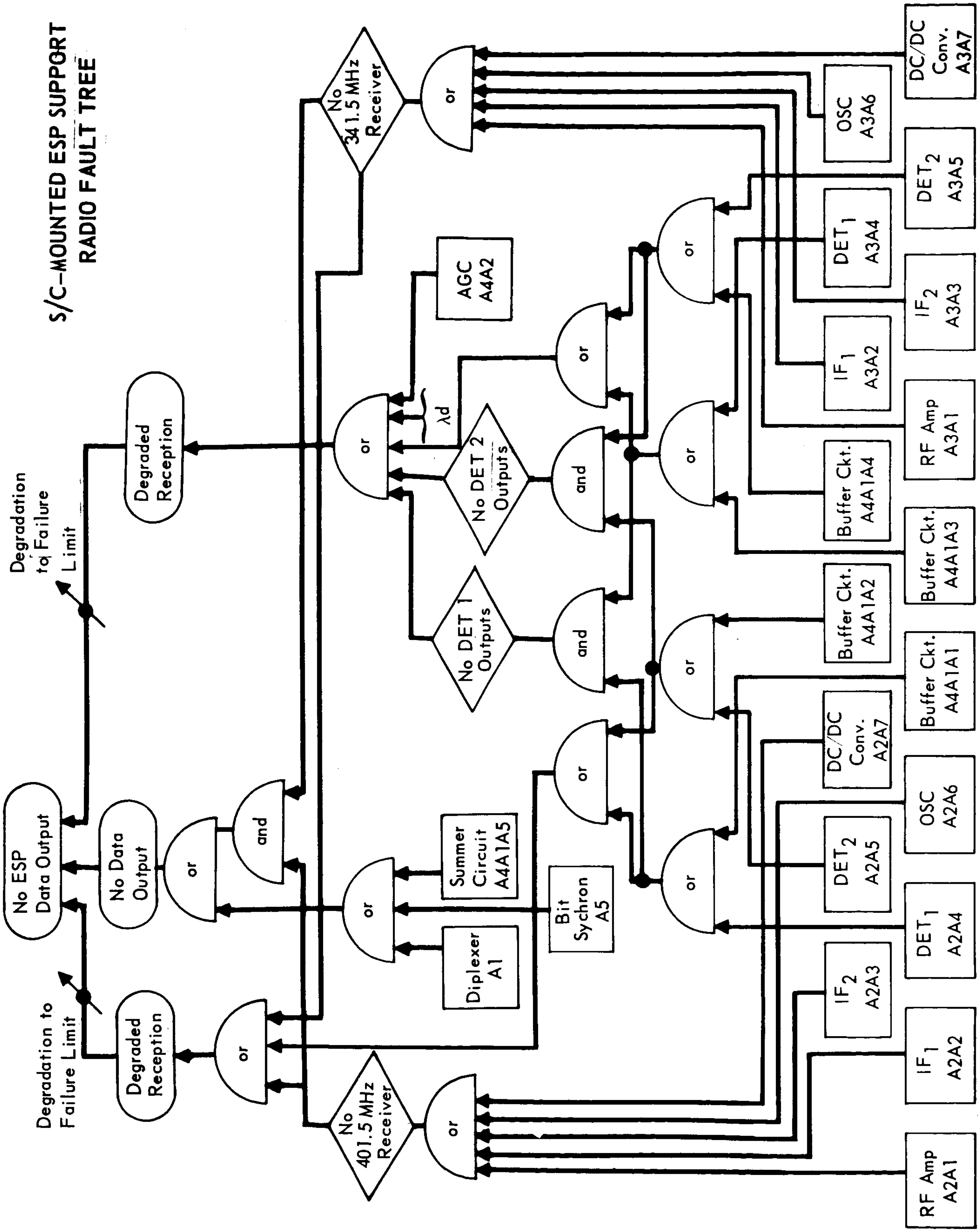

S/C-MOUNTED ESP SUPPORT RADIO FAULT TREE
No ESP Data Output
Degradation to Failure Limit
Degradation to Failure Limit
Degraded Reception
Degraded Reception
No Data Output
or
and
or
or
Diplexer A1
Summer Circuit A4A1A5
Bit Sychron A5
No 401.5 MHz Receiver
No 341.5 MHz Receiver
AGC A4A2
λd
No DET 1 Outputs
No DET 2 Outputs
and
and
or
RF Amp A2A1
IF1 A2A2
IF2 A2A3
DET1 A2A4
DET2 A2A5
OSC A2A6
DC/DC Conv. A2A7
Buffer Ckt. A4A1A1
Buffer Ckt. A4A1A2
Buffer Ckt. A4A1A3
Buffer Ckt. A4A1A4
RF Amp A3A1
IF1 A3A2
IF2 A3A3
DET1 A3A4
DET2 A3A5
OSC A3A6
DC/DC Conv. A3A7

Figure 5.2–14

- Degraded Reception - Both receivers operating but degraded in one or more parameters that effectively increase the bit error rate.
- Degraded Link - One of the two receivers failing to a "no output" resulting in a degraded signal to multipath interference noise ratio.
- No Data Output - Effective/catastrophic loss of the ESP to FSC data link due to a failure(s) that result in no FSC ESP Radio output. Alternate functional path is via the ESP to FSC link by data interleaving.

5.2.7.4 Complexity Estimate - The ESP Radio is estimated to have a complexity in the order of 910 piece parts plus hardware and connection system. Figure 5.2-15 is a table that presents a breakdown of the subsystem piece parts by category.

The critical application parts are the crystals and their mounting. The RF mixer diodes are not considered a critical application at these UHF frequencies.

5.2.7.5 Safety Considerations - No special safety considerations are denoted since the voltages of the units are in the 28 Vdc region.

5.2.8 Test - Testing will be conducted as indicated in the test matrix of Figure 5.2-16 in accordance with the Integrated Test Plan.

All ground testing shall be performed with the use of Operation Support Equipment (OSE). In-flight checkout (pre-separation) is initiated by the appropriate Flight Capsule test programmer upon command from the DSIF and monitored through the spacecraft telemetry system.

5.2.9 Development Status - Most of the RF components and IF amplifier circuitry are similar to those used in the receiver section of the Mariner 69 transponder. The video data circuit components are mostly integrated circuits. The frequency diplexer and crystal filters are virtually "off-the-shelf" items. No long lead time developments will be required.

SPACECRAFT RADIO SS PARTS COUNT ESTIMATE

PART TYPE	FR λ	SINGLE RECEIVER n_1	DUAL RECEIVER $2n_1$	DATA COMB n_2	BIT SYNC n_3	QUANTITY TOTAL ϵn	PRODUCT TOTAL $\epsilon n \lambda$
Capacitor							
Ceramic/Glass	1	76	152	4	38	194	194
Solid Tantalum	2	4	8		2	10	20
Variable Piston	10	21	42			42	420
Feedthrough	5	35	70			70	350
Diodes							
General Purpose	1.1	15	30	4	12	46	51
Zener	10	5	10		2	12	120
Mixer, RF	50	4	8			8	400
Resistors							
Metal Film	0.3	27	54	14	13	81	24
Carbon Comp.	0.1	93	186		14	200	20
Transistors							
General Purpose	5	24	48		26	74	370
Power	50	1	2			2	100
Integrated Circuits	10				23	23	230
Linear Op. Amp.	30	1	2			2	60
Inductors							
RF Chokes	7.5	33	66			66	495
LV Coils/Induct.	10	3	6			6	60
Transformers							
RF/IF Signal	5	16	32			32	160
LV/Power	5	5	10			10	50
Crystals	65	1	2			2	130
Decoupling Filters	5	12	24			24	120
Crystal BPF	150	2	4			4	600
Diplexer	10		1			1	10
Antenna and Cabling	48	1				1	48
Total		379	757	22	130	910	4032

Figure 5.2-15

TEST MATRIX S/C-MTD ESP SUPPORT RADIO

RADIO	ACCY REQ.		0.5 dB	2%	1 x 10−8	2%	1%	2%	1 dB	5%	1 dB		2%
	TEST	Non-Operative Test	Threshold Signal	AGC Voltage	Local Oscillator Frequency	Mixer Current	Power Supply Voltage	DC Current Drain	Dynamic Range	Bandwidth	Image Rejection		Video Output Monitor
Telemetry				X		X	X	X					X
System Test			X	X	X	X	X	X	X	X	X		X
Pre-Launch			X	X		X	X	X					X
In-Flight Checkout				X		X	X	X					X

Figure 5.2-16

REFERENCES

Section 5.1.2

Ref. 5.1.2-1 - 1973 VOYAGER Capsule Systems Constraints and Requirements Document, Jet Propulsion Laboratory, 1 January 1967 and 18 May 1967, California Institute of Technology.

SECTION 6

ENTRY SCIENCE PACKAGE ANTENNA SUBSYSTEM

The ESP antenna subsystem supports RF transmission from the ESP radio subsystem to the FSC mounted radio subsystem high rate equipment. The antenna subsystem includes provisions for radio subsystem testing (except for the ESP receiving antenna) in the pre-flight and pre-separation sequences, as shown in Figure 6-1. A summary of component characteristics is presented in Figure 6-2. All subsystem components are passive 50 ohm devices.

6.1 EQUIPMENT IDENTIFICATION AND USAGE

6.1.1 ESP Transmitting Antenna - The only subsystem component which will be mounted on the Capsule Lander, the ESP transmitting antenna radiates 90 watts of RF power from the ESP radio subsystem to the FSC mounted equipment of the ESP antenna subsystem. The antenna radiates whenever the ESP radio subsystem is transmitting, during pre-flight and in-flight testing, and from CB entry until landing.

6.1.2 ESP Test Antenna - Provided as part of the ESP data path during test, the ESP test antenna receives a portion of the RF power radiated from the transmitting antenna.

6.1.3 ESP Test Attenuator - Included to provide further attenuation during test, the attenuator provides 60 dB attenuation, with a dissipation rating of 10 watts.

6.1.4 ESP Directional Coupler - The directional coupler is common to the test and operational circuitry, and provides 50 dB isolation between the two paths.

6.1.5 ESP Receiving Antenna - The ESP receiving antenna, a LHC polarized helix, will be mounted on a mast which extends from the FSC. The antenna pattern encompasses the relative look angles to the CB during the period from entry until landing and receives ESP radiation at 341.5 MHz and 401.5 MHz.

6.2 DESIGN REQUIREMENTS AND CONSTRAINTS

6.2.1 ESP Transmitting Antenna

6.2.1.1 Pattern Coverage - Adequate pattern coverage is required to insure sufficient power radiated toward the FSC from entry until landing. The coverage will take account of the effects of varied CB attitude and CB/FSC geometry during the above period.

6.2.1.2 Multipath - The antenna pattern must be shaped to minimize multipath interference,with sharp pattern rolloff and low backlobes.

ENTRY SCIENCE PACKAGE ANTENNA SUBSYSTEM

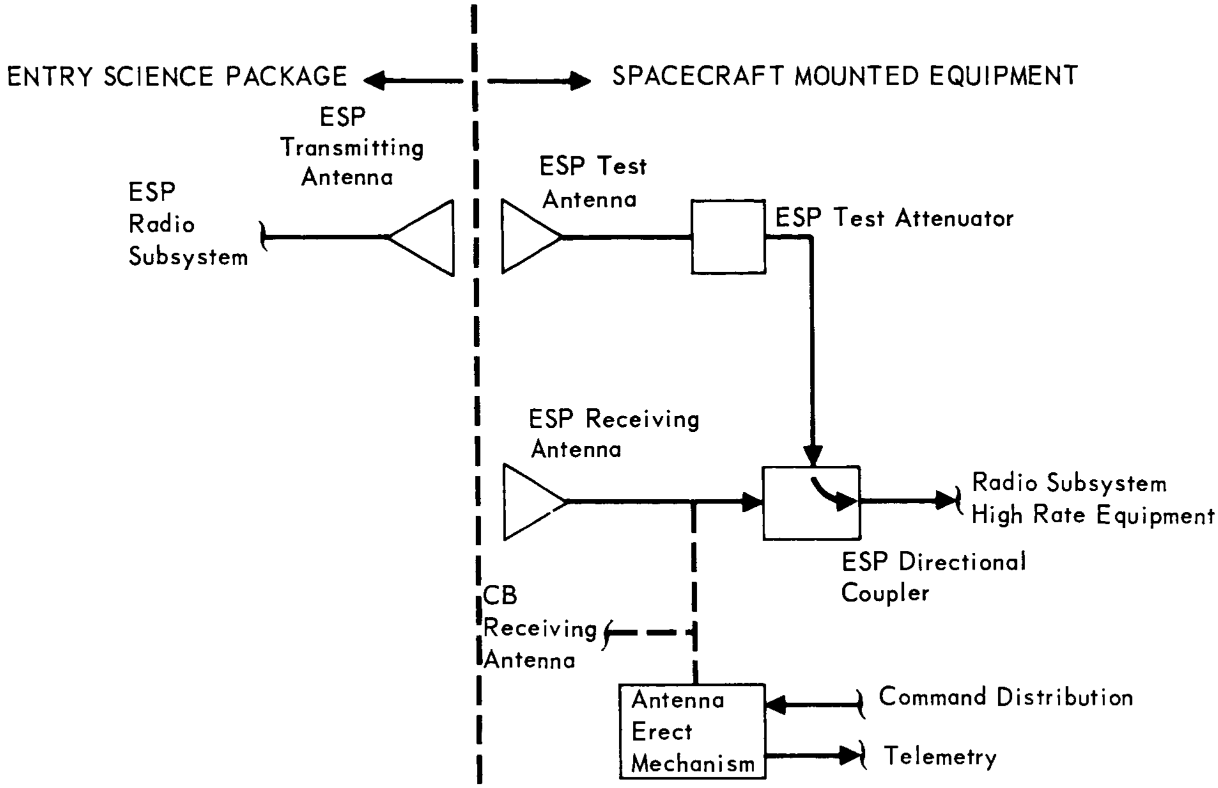


Figure 6-1

ENTRY SCIENCE PACKAGE ANTENNA SUBSYSTEM SUMMARY

COMPONENT	LOCATION	VOLUME	WEIGHT	POWER	GAIN	POLARIZATION
ESP Transmitting Antenna	Capsule Lander	1240 cu in.	3 lb	80 W	5.1 dB	LHC
ESP Test Antenna	Capsule Adapter	603 cu in.	7 lb	80 W*	–	LHC
ESP Test Attenuator	Flight Spacecraft	20 cu in.	1/2 lb	10 W*	–	–
ESP Directional Coupler	Flight Spacecraft	30 cu in.	3/4 lb	–	–	–
ESP Receiving Antenna	Flight Spacecraft**	2620 cu in.	1 lb	–	9.9 dB	LHC

* Dissipation Capability

** Mast Mounted

Figure 6-2

6.2.1.3 Polarization - The antenna must be circularly polarized to account for variations in the relative attitudes of the CB and FSC. The polarization is left hand circular (LHC) to minimize coupling with the CB antenna subsystem, which operates with RHC polarization in the same frequency band.

6.2.1.4 Bandwidth - The antenna must operate at 341.5 MHz and 401.5 MHz.

6.2.1.5 Temperature - The antenna must perform before, during, and after the high temperature period of entry. An antenna surface temperature of 650^{o}F is expected, with a conservative upper limit of 1100^{o}F.

6.2.2 ESP Receiving Antenna - The receiving antenna must maintain pattern coverage to the CB, within the 3 dB beamwidth, during the period from entry until landing. The antenna is LHC polarized to optimize reception of radiation from the LHC polarized ESP transmitting antenna, and reject radiation from the RHC polarized CB transmitting antenna.

6.3 PHYSICAL CHARACTERISTICS

6.3.1 ESP Transmitting Antenna - The transmitting antenna is a cavity backed Archimedes spiral, seven inches deep, with a fifteen inch diameter. Volume and weight are shown in Figure 6-2. Location on the Capsule Lander is shown in Figure 6-3. The antenna is raised above the surrounding structure to protrude through the thermal curtain with a clear field of view $\pm$60 degrees off the -Z axis.

6.3.2 ESP Test Antenna - The test antenna is a cylinder, three inches high and sixteen inches in diameter, lined with RF absorber (Emerson and Cuming NZ-2 satisfies the temperature and free space environment requirements) and dielectric.

6.3.3 ESP Test Attenuator - The coaxial attenuator will be mounted in the FSC, with volume and weight requirements as shown in Figure 6-2.

6.3.4 ESP Directional Coupler - The directional coupler with approximate dimensions of six inches by five inches by one inch will be mounted in the FSC.

6.3.5 ESP Receiving Antenna - The ESP receiving antenna is a LHC helix, 30 inches long and 10 inches in diameter with 4.5 turns and a 12.5^{o} pitch angle. Weight and volume are shown in Figure 6-2. The antenna will be mounted on a common mast with the CB receiving antenna, located as shown in Figure 6-4, and pointed to accommodate line of sight variations for the predicted entry through landing sequence.

6.4 PERFORMANCE CHARACTERISTICS

6.4.1 ESP Transmitting Antenna - Model antenna tests have been conducted to determine the antenna configuration which best satisfies the pattern coverage and

ESP TRANSMITTING ANTENNA LOCATION

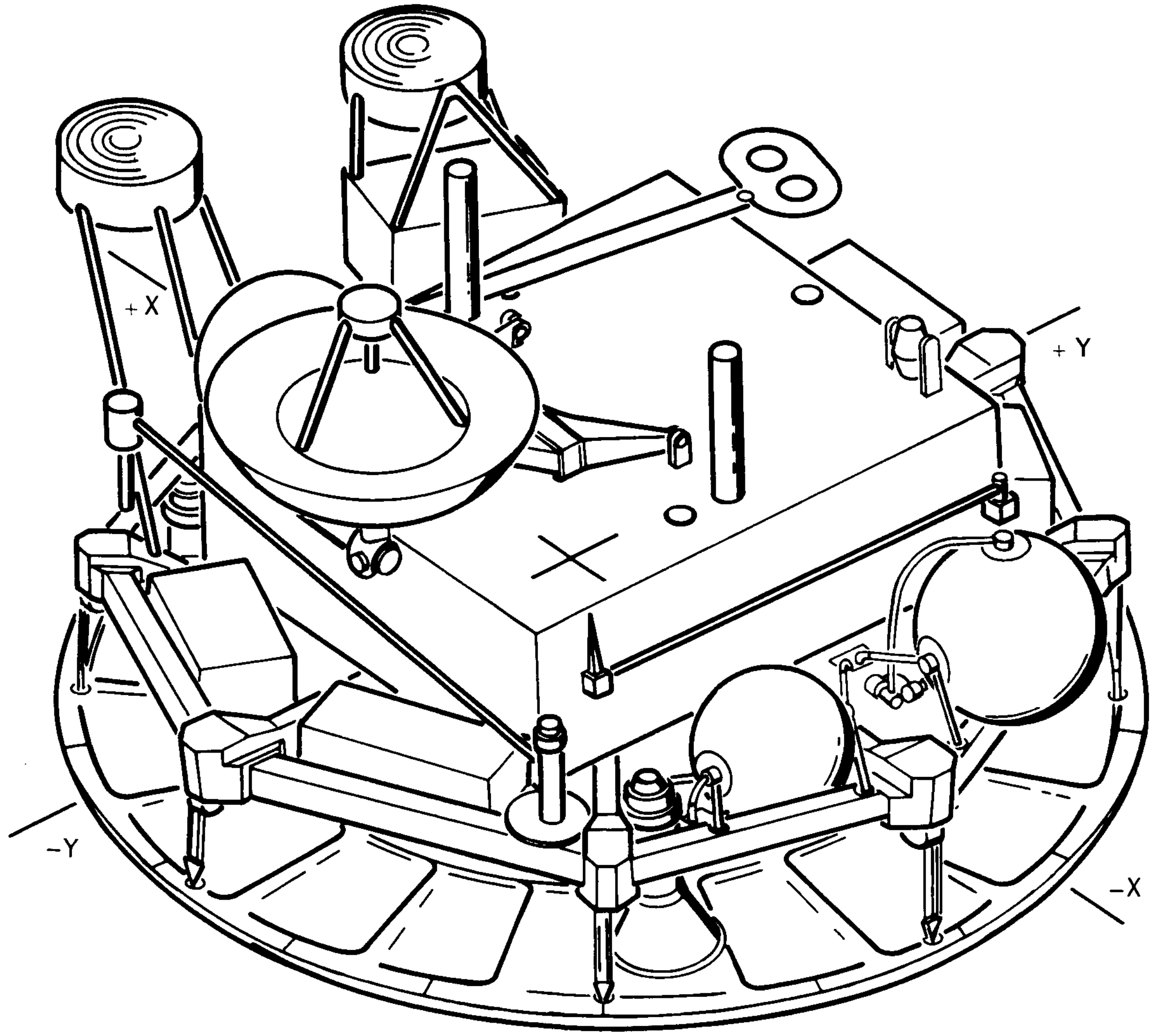


Figure 6-3

ENTRY SCIENCE PACKAGE RECEIVING ANTENNA LOCATION

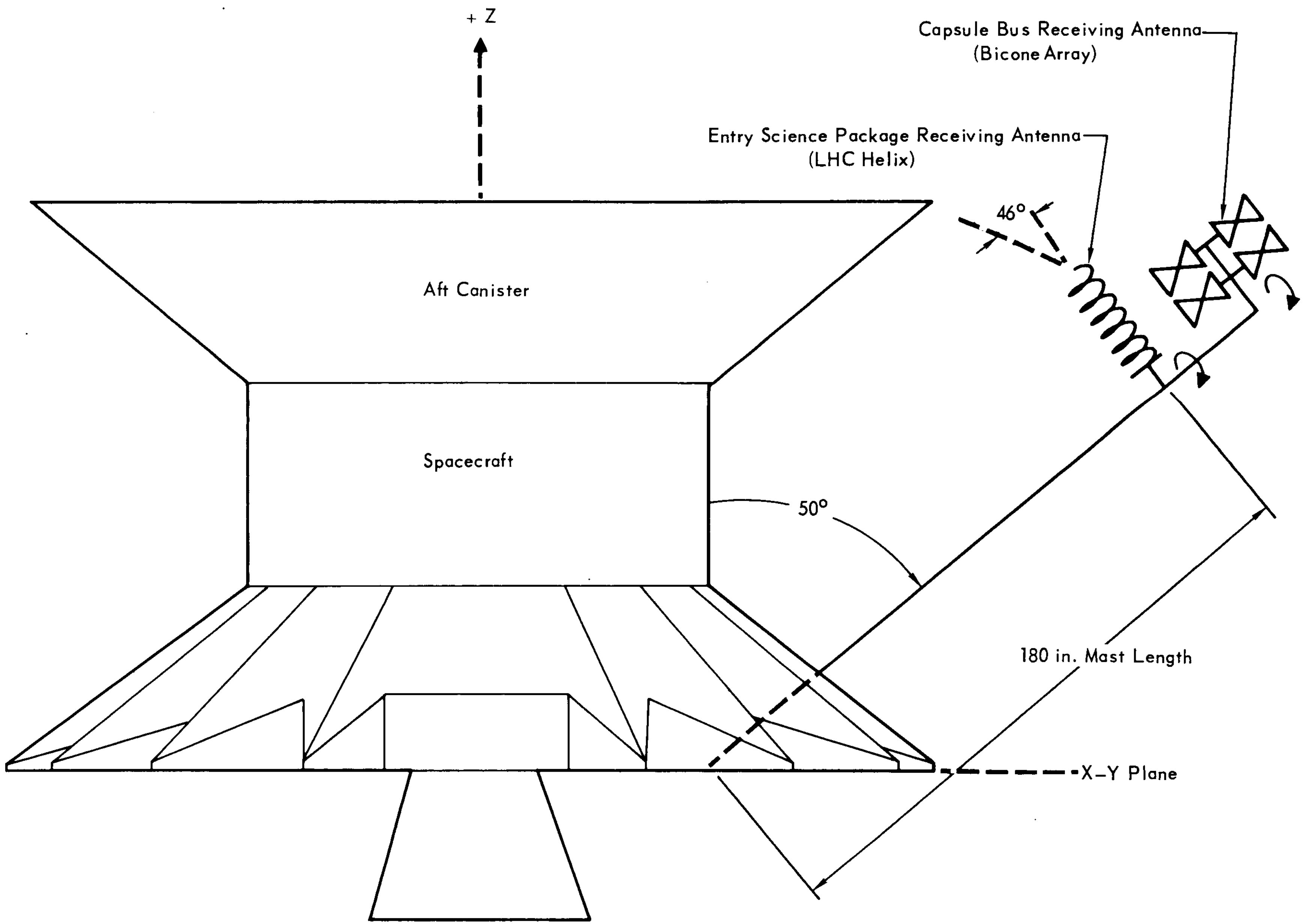


Figure 6–4

multipath requirements. The most favorable antenna, an Archimedes spiral above a cavity, presents a pattern as shown in Figure 6-5. Maximum gain is approximately 5.1 dB (70% efficient) and the beamwidth is approximately 95 degrees. Average gain slope for angles beyond 70 degrees off axis, is 0.73 dB per degree. Backlobe radiation is approximately 26 dB below maximum gain. The on-axis axial ratio is less than 1 dB, with LHC polarization. As shown, a conic section 55 degrees off axis exhibits less than 1.5 dB variation with a symmetrical ground plume. Model tests were performed at 1200 MHz.

6.4.2 ESP Test Antenna - Refer to Figures 6-2 and 6-6.

6.4.3 ESP Test Attenuator - Refer to Figures 6-2 and 6-6.

6.4.4 ESP Directional Coupler - Refer to Figures 6-2 and 6-6.

6.4.5 ESP Receiving Antenna - The ESP receiving antenna performance characteristics are predictable on the basis of extensive documentation of helical antennas. The gain is approximately 9.9 dB for 70 percent efficiency, with a 55 degree beamwidth symmetrical pattern, and an on-axis axial ratio of approximately 0.5 dB.

6.5 RELIABILITY AND SAFETY CONSIDERATIONS

6.5.1 Reliability - The antennas present single point failure modes in the link between the ESP radio subsystem and the FSC-mounted radio subsystem high rate equipment, for television transmission. However, a functional backup exists in the CB antenna subsystem for all other ESP data, due to interleaving between the CB and ESP data link. The most probable failure mechanism for the antennas is the result of physical damage to the connectors, feed points or elements, which would probably occur during handling or installation, and would be evident upon visual inspection or subsystem testing prior to sterilization.

The antenna positioning mechanism will be the most simple, reliable design possible, and will be compatible with the FSC configuration.

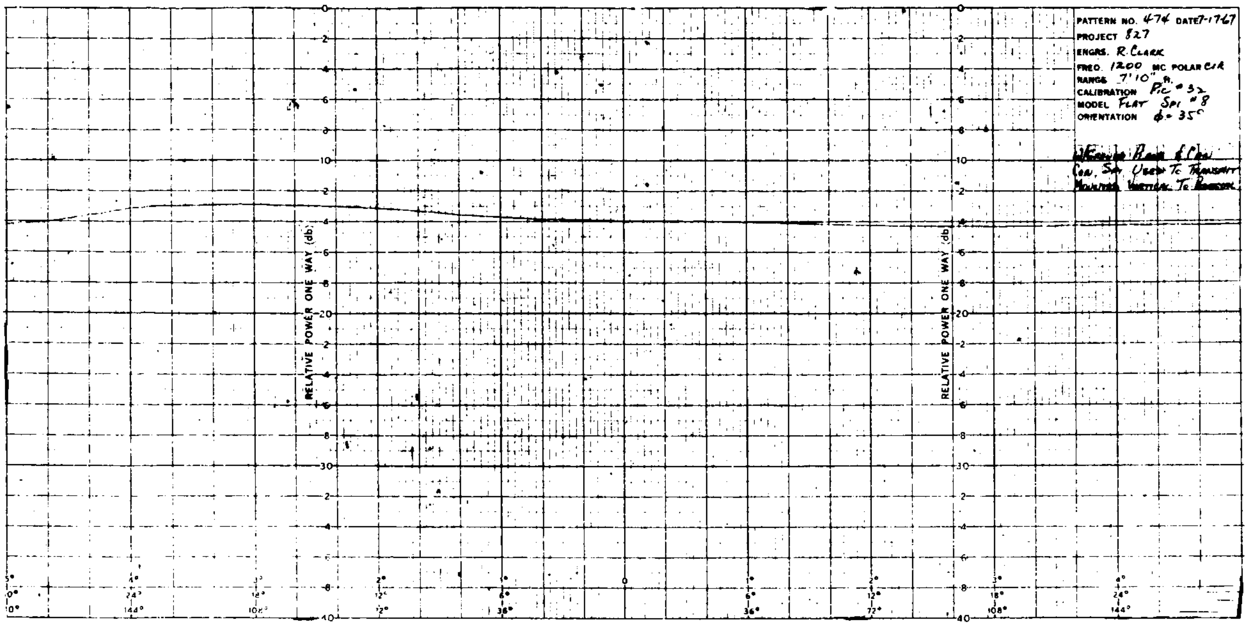
6.5.2 Safety - The ESP transmitting antenna will radiate approximately 90 watts of RF power during tests after installation. The antenna is included as an operating element of the test path in order to eliminate the need for RF switches. To eliminate stray radiation, the ESP test antenna will cover the ESP transmitting antenna until CB separation.

6.6 TEST

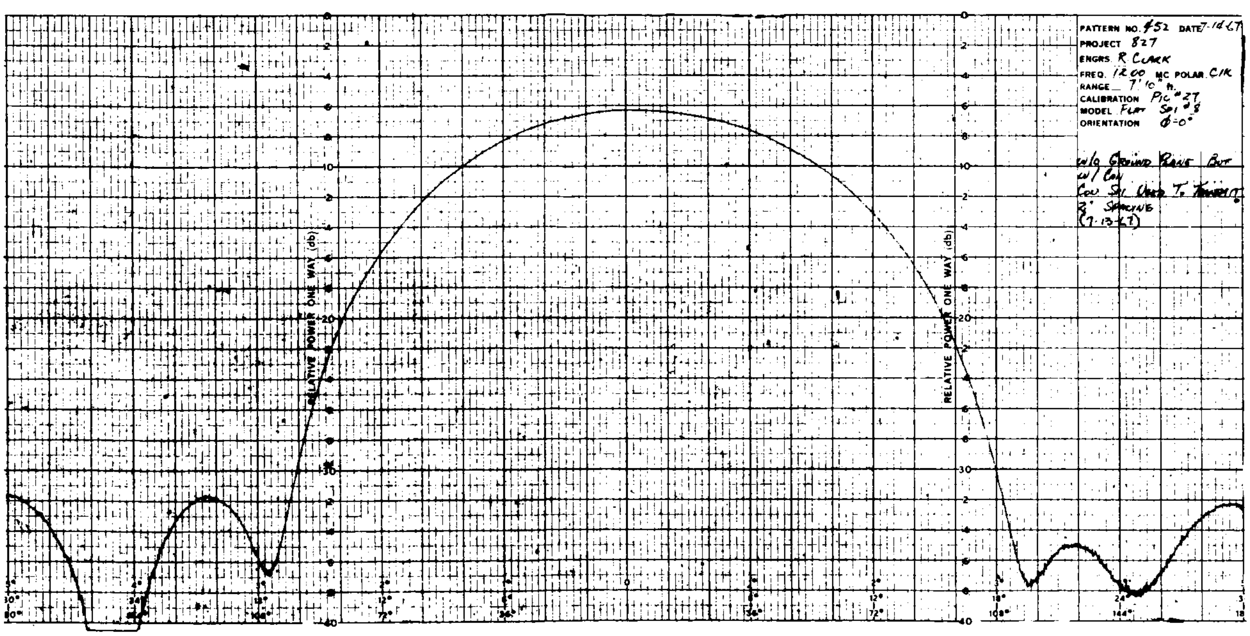
6.6.1 ESP Transmitting Antenna - The transmitting antenna will be tested for pattern coverage and impedance characteristics after delivery. Impedance characteristics will be measured during and after life cycling to insure proper

ENTRY SCIENCE PACKAGE TRANSMITTING ANTENNA

CONIC SECTION 55° OFF AXIS
(1/3 SCALE MODEL)



RADIATION PATTERN
(1/3 SCALE MODEL)



6-7

POWER FLOW THROUGH ENTRY SCIENCE PACKAGE ANTENNA SUBSYSTEM DURING TEST

COMPONENT	POWER IN	POWER OUT	LOSS MECHANISM
ESP Transmitting Antenna	49.5 dBm	49.5 dBm	Radiation
ESP Test Antenna	49.5 dBm	+6.3 dBm	Absorption & Decoupling
ESP Test Attenuator	+6.3 dBm	−53.7 dBm	Dissipation
ESP Directional Coupler	−53.7 dBm	−103.7 dBm	Reflection

FSC Mounted Radio Subsystem High Rate Equipment Dynamic Range; −109.7 dBm to +20 dBm

Figure 6-6

performance subsequent to sterilization and during entry heating. Breakdown characteristics will be determined in the spectrum of Mars atmospheres. Explicit measurement of antenna performance will not take place during subsystem testing. However, transmitting antenna performance will be implied in forward and reflected power measurements made between the transmitters and diplexer of the ESP radio subsystem.

6.6.2 ESP Test Antenna - Testing before installation will include satisfactory separation from the ESP transmitting antenna and survival of the sterilization environment. The antenna must retain a RF seal around the transmitting antenna after sterilization, and must release at CB separation.

6.7 DEVELOPMENT REQUIREMENTS

6.7.1 ESP Transmitting Antenna - High temperature spirals have been built for S-band operation. Development will be required to obtain high temperature performance at UHF, due to increased antenna size.

6.7.2 ESP Test Antenna - The test antenna must be designed to satisfy the requirements peculiar to the application. However, no material developments are required. Development will be required to arrive at the optimum configuration.

SECTION 7

ELECTRICAL POWER SUBSYSTEM

The Electrical Power Subsystem provides the following:

a. Power for operation of the Entry Science Package (ESP) cruise commutator part of the Telemetry Subsystem, and for instrumentation when Flight Spacecraft (FSC) power is not available to the Flight Capsule (FC) during cruise and Mars orbit operation.

b. Power for operation of ESP equipment during a portion of the preseparation checkout operations and during a period from approximately one hour prior to separation of the FC from the FSC until 10 minutes after landing.

c. Control of power distributed to the ESP equipment.

7.1 EQUIPMENT IDENTIFICATION AND USAGE - The subsystem contains a battery, battery charger, and power switching and logic unit. A schematic block diagram of the subsystem is shown in Figure 7-1.

7.1.1 Battery - The ESP battery is a sealed silver zinc battery containing 20 eight ampere-hour cells connected in series. This battery is designed to provide a minimum of four complete charge-discharge cycles and is designed to be sterilized in a discharge condition prior to the formation charge. It provides power for operation of ESP equipment during cruise when FSC power is not available and during descent of the FC to the Mars surface.

7.1.2 Battery Charger - The battery charger is a two step float charger. It maintains the battery in a charged condition during cruise and after FC preseparation checkout. The charger has two modes of operation. In the first mode, it charges the battery at a constant potential of 1.98 volts $\pm$ 0.01 volts/cell with current limiting at 0.16 amps (C/50 rate). Charging current is sensed and when this reduces to 0.08 amps (C/100 rate), the battery is put on float charge at 1.87 volts $\pm$ 0.01 volts/cell. Each time power is removed from the battery charger and then reapplied, the charger reverts to the first operating mode.

7.1.3 Power Switching and Logic Unit - The power switching and logic unit (PS&L) in the ESP is a single unit. It provides voltage sensing, transfer to internal power when FSC power to the FC is turned off, and transfer to external power when FSC power is turned on. It provides ESP battery failure sensing and redundant mode switching, and control of power to the ESP equipment. The PS&L distributes power from the distribution bus to the ESP equipment. Overload-fault sensing devices protect the bus from load circuit failures. These devices actuate the equipment control relay turning off the equipment. The PS&L also contains a voltage sensor

ENTRY SCIENCE PACKAGE ELECTRICAL POWER SUBSYSTEM BLOCK DIAGRAM

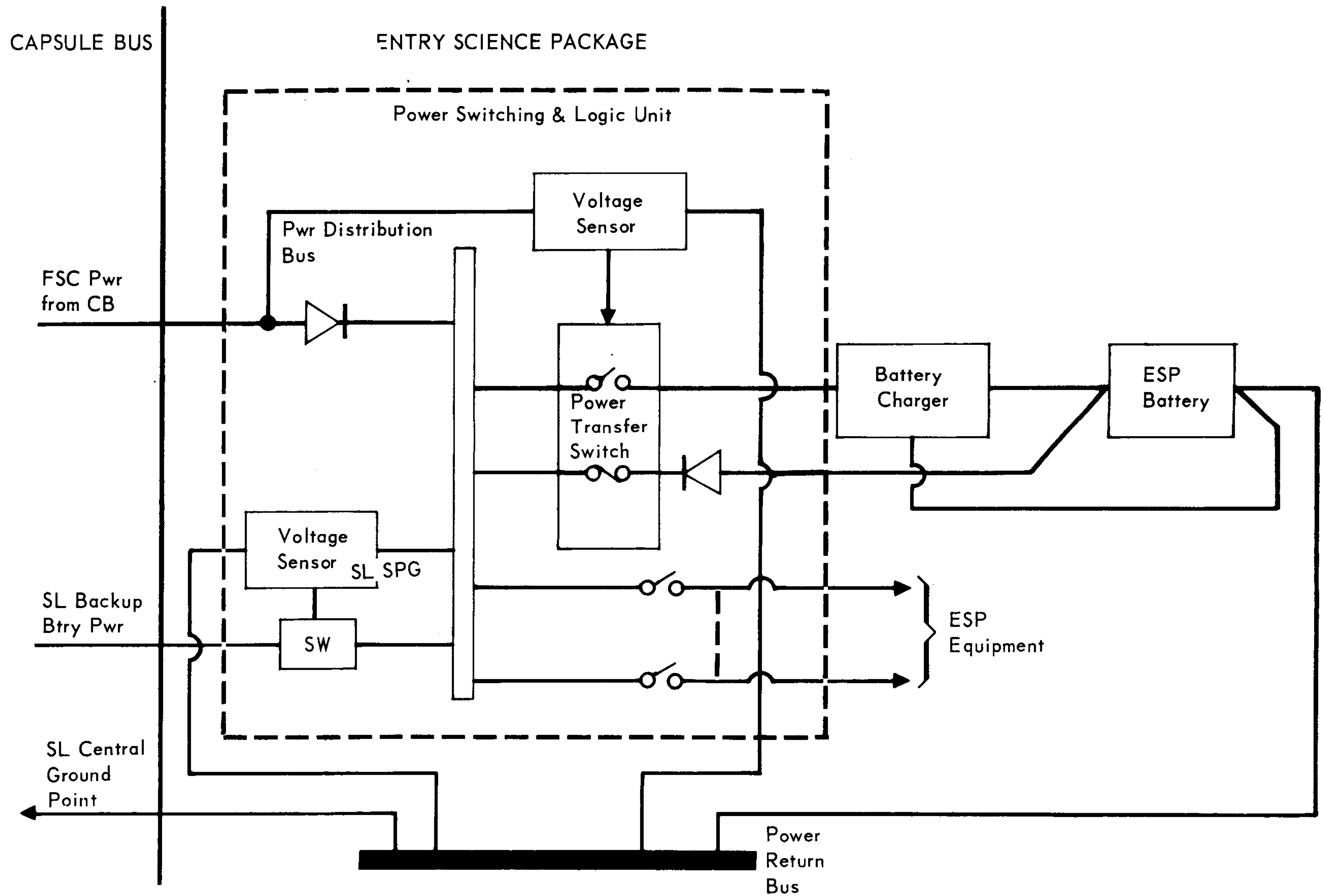


Figure 7–1

which senses loss of power from the ESP battery during descent and interconnects a battery in the SL to the power distribution bus during descent of the FC to the Mars surface.

7.2 DESIGN REQUIREMENTS AND CONSTRAINTS - The primary requirements and constraints influencing the design of the Entry Science Package Electrical Power Subsystem are the following:

a. The subsystem must provide approximately 345 watts of power to the ESP subsystems during descent and landing operations and 230 watt-hours of electrical energy.

b. It must provide approximately 10 watts of power for operation of telemetry, instrumentation, and thermal control heaters during cruise when FSC power is not available to the FC.

c. It must distribute FSC power to the operating ESP equipment during cruise when FSC power is available to the FC.

d. It must provide power switching devices for controlling the subsystem equipment in the ESP.

The electrical power requirements of the ESP equipment are shown in Figure 7-2.

7.3 PHYSICAL CHARACTERISTICS - The ESP Electrical Power Subsystem equipment total weight is 22.5 pounds and it occupies a volume of 396 cubic inches. The physical characteristics of the equipment are shown in Figure 7-3.

7.4 OPERATION DESCRIPTION - The ESP Electrical Power Subsystem mission sequence is shown in Figure 7-4. The following is a description of the subsystem operation by mission mode.

7.4.1 Prelaunch and Launch - The ESP battery is charged prior to launch using the on-board battery charger. During launch, power is supplied from the battery for in-flight monitoring and electrical heaters. Transfer to Flight Spacecraft (FSC) supplied power occurs after FSC solar cell panel deployment, with less than 20 watt hours having been removed from the battery. The battery charger will begin charging the battery. Transfer of the ESP systems to FSC power is controlled by a power transfer switch in the ESP Power Switching And Logic Unit (PS&L). One command for activating this switch comes from the spacecraft. A voltage sensor in the PS&L sensing turn-on of spacecraft power, provides a redundant command to activate the power transfer switch.

7.4.2 Cruise - During cruise, the operating equipment derives its power from the spacecraft via the Capsule Bus. This power is provided to the power distribution

ENTRY SCIENCE PACKAGE ELECTRICAL POWER REQUIREMENTS

EQUIPMENT	CRUISE-WATTS	MARS DESCENT		TOTAL ENERGY
		WATTS	TIME (min)	
Radio		270	27	121.5
Telemetry	2	2	450	15
		1.7	27	0.77
Instrumentation	3	3	450	22.5
		7	27	3.15
Data Storage		8	27	3.6
Descent Vidicons		20	27	9.0
Pressure Transducers		1	27	0.45
Accelerometers		2	27	0.9
Temperature Probes		0.2	27	0.09
Mass Spectrometer		7	27	3.15
Power Switching & Logic Unit	2	2	450	15
Battery Charger	5			
Electrical Heater	3	3	450	22.5
Line Losses (6%)	0.72	19.5		12.3
Total	15.9	346.4		229.91

Figure 7-2

ELECTRICAL POWER SUBSYSTEM PHYSICAL CHARACTERISTICS

Battery	
Weight	15 lb
Size	4.5 in. x 5.5 in. x 7.5 in.
Quantity	1
Cell/Battery	20
Battery Charger	
Weight	1.5 lb
Size	2.5 in. x 3 in. x 4 in.
Quantity	1
Power Switching and Logic Unit	
Weight	6 lb
Size	4 in. x 5 in. x 9 in.
Quantity	1

Figure 7-3

ENTRY SCIENCE PACKAGE ELECTRICAL POWER MISSION SEQUENCE

Mission Phase	Duration	Energy Source	Equipment operating
Prelaunch	3 days	OSE	Battery chargers all other equipment tested
Launch	3 hours	Battery	Telemetry equipment instrumentation
Transit	7 months	FSC	Battery chargers telemetry equipment, instrumentation
Trajectory Corrections	10 hours (Total)	Battery	Telemetry equipment instrumentation
Preseparation Checkout	4 hours	FSC	Equipment being tested telemetry equipment, instrumentation
Preseparation to Entry	7 hours	Battery	Telemetry equipment instrumentation
Entry and Terminal Deceleration	11.5 minutes	Battery	All
Post Landing	10 minutes	Battery	All

Figure 7-4

bus. Power is distributed from this bus to the operating equipment. The average power required during this period is approximately 15 watts.

7.4.3 Periods of High FSC Power Usage - When spacecraft power is turned off, a spacecraft command transfers the ESP to internal battery power. The voltage sensor provides a redundant command to accomplish this. The maximum amount of energy removed from the battery during any of these periods (3.5 hours assumed) is not expected to exceed 40 watt-hours. When spacecraft power again becomes available, a command transfers the ESP back to spacecraft power and puts the battery back on float charge. The voltage sensor again is a redundant path for accomplishing this.

7.4.4 Pre-separation Checkout - Pre-separation checkout will be performed using primarily FSC power. End to end tests will require more than 200 watts, however, and the CB test programmer will transfer the ESP to internal battery power in order to run these tests. The depth of discharge of the battery is expected to be less than 10%. After completion of the pre-separation checkout, the battery will be recharged for 9 hours.

7.4.5 Pre-separation to CB Shutdown - Approximately one hour prior to FC/FSC separation, the ESP is transferred to internal power by a command from the CB Sequencer & Timer (S&T) and a voltage sensor in the PS&L is enabled. Loss of battery power causes the voltage sensor to activate a switch allowing power from the SL to be made available to the ESP.

7.5 PERFORMANCE CHARACTERISTICS - The performance characteristics of the power subsystem equipment can be well defined except for the battery. Sterilizable, sealed, silver-zinc batteries have given satisfactory post sterilization performance. Limited demonstration testing has been accomplished, however, and therefore performance objectives presented here are necessarily conservative. The expected performance characteristics of the electrical power subsystem equipment are shown in Figure 7-5.

The battery terminal voltage will be between the limits of 25 and 37 vdc. The ESP equipment will be required to operate between 23.5 and 33.5 volts except for the telemetry equipment and instrumentation power supply operating during cruise which present a low load to a fully charged battery.

7.6 INTERFACE DEFINITION - The Entry Science Package Electrical Power Subsystem has interfaces with the Capsule Bus and all ESP subsystems. These interfaces are shown in Figure 7-6. Within the ESP, the interfaces with the subsystem equipment consist of latching relays within the PS&L which turn the using subsystem equipment on and off. These relays are activated from the CB Sequencer & Timer during descent

POWER SUBSYSTEM PERFORMANCE CHARACTERISTICS

Battery
- Type – Sealed AgZn
- Nominal Cell Capacity (to 1.25 Volts/Cell) = 8 amp hr
- Number of Cells/Battery = 20
- Battery Energy = 230 watt-hr
- Battery watt-hr per lb = 15
- Wet Stand Life = 12 months
- Battery Voltage Regulation = 25 to 34 volts-nominal
  37.2 volts-charged, open circuit
- Operating Temperature Environment
  - On Float Charge = 0°F to 60°F
  - Nominal Use Environment = 50°F to 120°
- Cycle Life – 4 Discharge-Charge Cycles to 100% of Rated Capacity

Battery Charger
- Type – Float Charger
  - Mode No. 1 – Constant Potential – 39.6 ± 0.2 vdc
    Current Limiting at 0.16 amps ± 5%
  - Mode No. 2 – Constant Potential – 37.4 ± 0.2 vdc
  - Mode Charge (1 to 2) Trip Point = 0.8 amps ± 5%
  - Normal Input Power (Mode No. 2) = 5 watts
  - Operating Temperature Range = –65°F to + 165°F

Figure 7–5

ESP ELECTRICAL POWER SUBSYSTEM INTERFACE BLOCK DIAGRAM

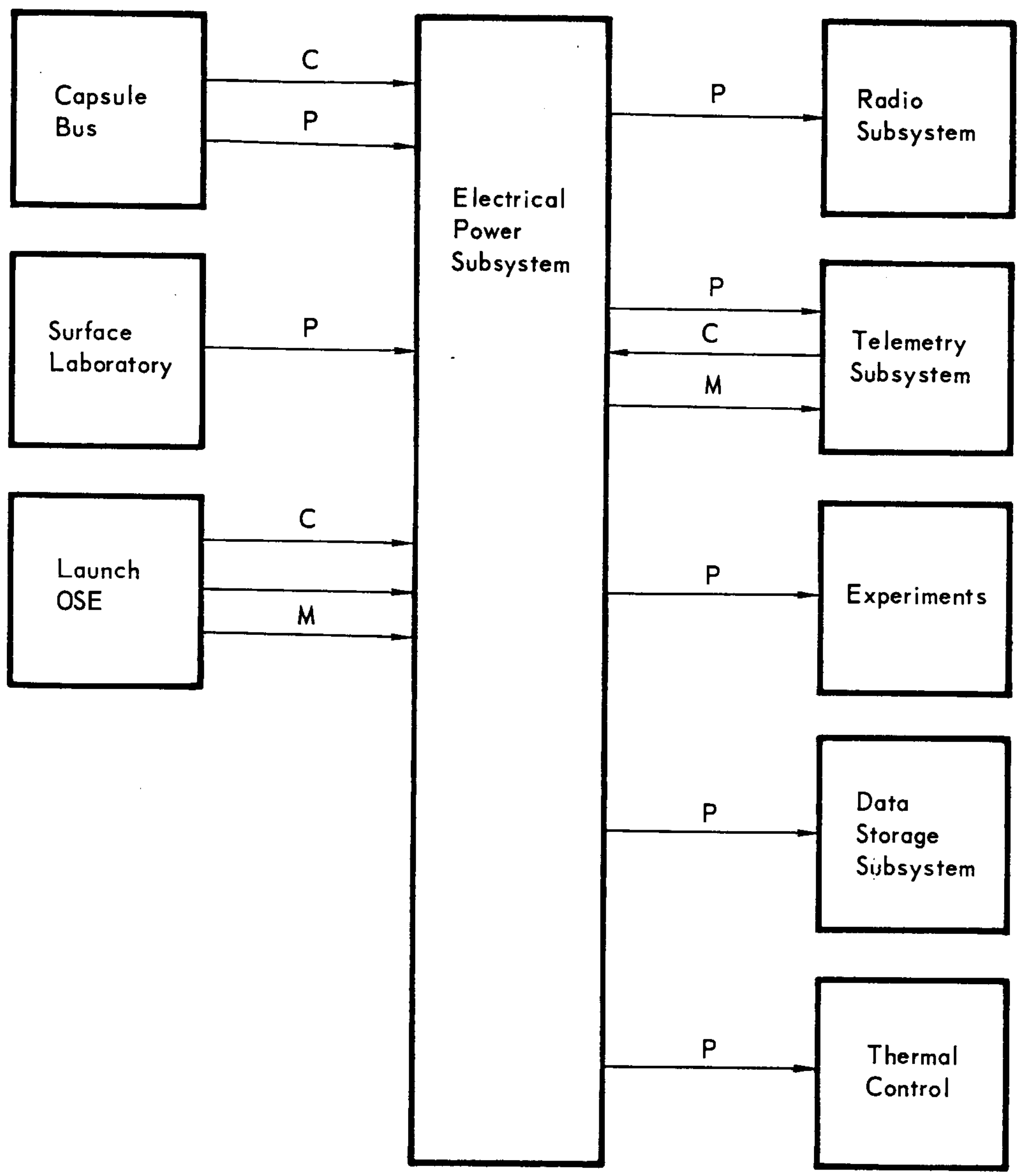


Figure 7–6

to the Mars surface, and from the Test Programmer in the Capsule Bus during pre-separation checkout. The in-flight commands to the subsystem are shown in Figure 7-7. The in-flight monitoring requirements of the subsystem are shown in Figure 7-8.

The electrical power return bus of the Entry Science Package is not grounded to the structure of the ESP. The return bus is connected to the power return bus in the Capsule Bus which is returned to the single point structural ground in the Surface Laboratory.

7.7 RELIABILITY CONSIDERATIONS - The Electrical Power Subsystem reliability considerations center about assurance that power for ESP equipment operation is available during the mission. This assurance comes through conservative design, simplicity, and redundancy. With the exception of the necessity of developing a sterilizable battery, the remainder of the elements involve present-day hardware and design technique.

7.7.1 Operational Reliability - The system is configured simply with operational reliability provisions as follows:

a. Power transfer switch command function is assured via functional redundancy means; one mode is the automatic ESP self-contained bus voltage sensor while the other is the spacecraft command means.

b. Battery power is assured by several means, all of which enhance mission success. The ESP battery operational life will be considerably strengthened by the charge control method to be utilized. The float charging concept or charge control method has shown most promising results during tests at Goddard Space Flight Center. The float charge technique has thus far indicated (1) essential elimination of cell unbalance problems during cycling, (2) contributions to preventing premature catastrophic failure during cyclic life, and (3) retardation of silver migration and zinc dendritic growth. Although test data is limited, this means was chosen for the VOYAGER program in consideration of the extended cruise period (7 months) and intermittent cycle duty.

   In addition to the above battery charge control method, the ESP battery power is backed up by SL batteries.

7.7.2 Failue Mode, Effect, and Criticality Analysis (FMECA) - A failure mode, effect, and criticality analysis was conducted for the Electrical Power System and results are presented in Figure 7-9. Each failure mode is categorized according to the effect on the following mission objectives:

a. Achievement of Flight Capsule landing

ELECTRICAL POWER SUBSYSTEM COMMAND INTERFACE LIST

FUNCTION \ COMMAND SOURCE	CB S&T	FSC-CC&S	CAPSULE BUS COMMAND DECODER
Operate PS&L Power Transfer Switch	√	√	
Enable Voltage Sensor No. 2	√		√
Turn Battery Charger On or Off			√
Turn Descent Equipment On	√		
Turn Off ESP Equipment	√		

Figure 7-7

ELECTRICAL POWER SUBSYSTEM – IN FLIGHT DATA REQUIREMENTS

EQUIPMENT	MISSION PHASE	SAMPLE FREQUENCY SAMPLES/SEC	PURPOSE OF TEST	CORRECTIVE ACTION
Battery Voltage	Cruise	0.01	Verify Charging	Command Charger On or Off
	Preseparation	0.01	Monitor Performance	Command Redundant SL Power
	Descent	0.01	Monitor Performance	None
Battery Current	Cruise	0.01	Monitor Performance	Enable Voltage Sensor No. 2
	Preseparation & Descent	0.01	Monitor Performance	None
Battery Temperature	Cruise	0.01	Diagnostic Data	Reduce or Increase Heat Input
	Preseparation & Descent	0.01	Monitor Performance	Command Redundant SL Power
Battery Charge Current	Cruise	0.01	Monitor Performance	Command Charger On or Off
Power Transfer Switch Position	Cruise, Preseparation & Descent	0.01	Verify Operation	Command Again
Redundant Surface Lab Power Switch	Preseparation & Descent	0.01	Verify Transfer to SL Power	None

Figure 7-8

FAILURE MODE, EFFECT AND CRITICALITY ANALYSIS

ESP ELECTRICAL POWER

COMPONENT OR FUNCTION	FAILURE MODE	FAILURE EFFECT	FAILURE CATEGORY: LANDING	ENTRY SCIENCE	LANDED SURFACE	ENG DATA	REMARKS
PS&L Input Power Diode	Open	None	1	1	1	1	Loss of Spacecraft power input which is ESP Battery backed up for ESP functions and the ESP battery is backed up by SL.
Voltage Sensor #1 (S/C Power Loss)	Inoperative	None	1	1	1	1	Backed up by Spacecraft command for power transfer switching ESP battery on the line.
Voltage Sensor #2 (ESP Battery Loss)	Inoperative	None	1	1	1	1	No effect if ESP battery is operative. Loss of backup function.
Battery Float Charger	Inoperative	None	1	1	1	1	ESP battery energy decays normally at 4%/month rate. ESP battery backed up by SL.
Battery Main	Loss	None	1	1	1	1	ESP Battery backed up by SL. If SL Battery Charger Units full operative, back-up does not effect SL.

Failure Category Definition – 1 No effect on mission objectives
2 Degraded effect on mission objectives
3 Possible catastrophic effect on mission.

Figure 7-9

b. Performance of entry science experiments

c. Performance of landed science experiments

d. Retrieval of engineering data

The electrical power system FMECA analysis evidences only Category 1 failures.

7.7.3 Reliability Estimate - The functional relationship of components in the Electrical Power System is shown in the Reliability Diagram, Figure 7-10. The Reliability Estimate Summary, Figure 7-11, evidences the Electrical Power System calculated reliability of .9901.

The battery reliability estimate is based on the Surveyor silver-zinc battery cell test data utilizing 662 cells over a period of 383,000 hours. Testing was performed by simulating programed mission profile in an environmental chamber. The resultant cell mean time between failure was 239,392 hours. It was projected within the development span allowable for the VOYAGER mission, this failure rate could be improved an order of magnitude. Thus the failure rate utilized in the reliability estimate calculation was $0.4/\text{cell}/10^6$ hours with a resultant 20 cell battery failure rate of 8.0 per million hours. With similar development time the sterilizable silver-zinc battery may reach the same reliability attained by the Mariner silver-zinc battery, which demonstrated a mission probability of success of .998 @ 80% confidence level.

7.8 TESTING - Pre-launch testing of the ESP subsystem equipment will be accomplished using OSE power. The anticipated charge-discharge cycling of the flight battery is as follows:

a. Post Sterilization - one complete charge-discharge cycle.

b. Launch to Pre-separation - up to 6 discharge/charge cycles of less than 20% depth of discharge.

c. Mars Descent - complete discharge.

Battery performance testing prior to Flight Capsule terminal sterilization is not presently planned because some batteries under development must be sterilized prior to the formation charge.

7.9 DEVELOPMENT REQUIREMENTS - The Electrical Power Subsystem components are not long lead time development items except for Ag-Zn battery. Two 6-cell batteries have demonstrated the capability to survive the heat sterilization followed by the flight time on wet stand. In these tests the total heat sterilization time was less than the VOYAGER Type Approval Test requirements. Further battery development appears necessary in the area of:

a. Demonstration of the capability of sealed cells to repeatably survive the

ENTRY SCIENCE PACKAGE ELECTRICAL POWER RELIABILITY DIAGRAM

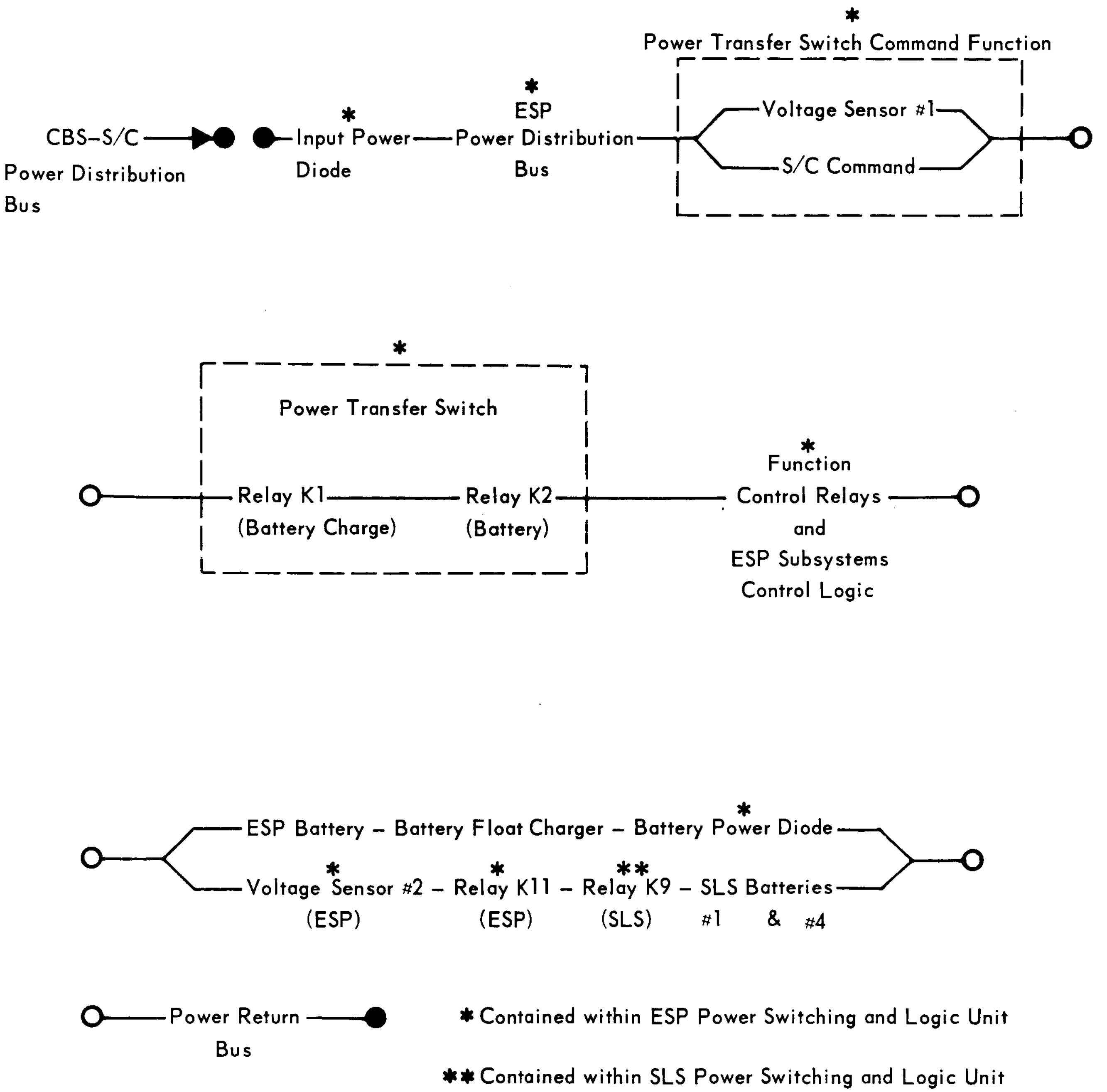


Figure 7–10

ESP ELECTRICAL POWER SUBSYSTEM RELIABILITY ESTIMATE SUMMARY

COMPONENT	tm (1)	λ (2)	$-\ln R \times 10^6$ (3)
PS&L			
Input Power Diode	5549	0.10	555
Power Transfer Command Function – Redundant	5549	–	≈ 0
Power Transfer Switch			
Relay K1 (Battery Charger)	5549	0·40	2220
Relay K2 (Battery)	136	0.40	54
Battery Power Diode	136	0.10	14
Function Control Relays			
K3 thru K10 Subsystems	124/Relay	0.40/Relay	400
Main Battery Power – Redundant	5597	–	6652
			9895 & R = .9901

(1) tm = Modified time factor x time (hours).
(2) λ = Failure per million hours
(3) $-\ln R \times 10^6 = \text{tm}\ \lambda$

Figure 7–11

sterilization environment without cell case rupture.

b. Demonstration of the capability of sealed cells to survive heat sterilization followed by 12 month wet stand and then give rated performance.

SECTION 8

STRUCTURAL/MECHANICAL SUBSYSTEM

8.1 EQUIPMENT IDENTIFICATION AND USAGE - The structural/mechanical subsystem provides the physical elements in the Flight Capsule for the support, alignment, protection, and separation of the components of the ESP Subsystem from the time of the initial installation of the ESP components until completion of their assigned functions in the total mission. The subsystem elements must be designed to withstand without failure the greatest anticipated loads to which they may be subjected. They must be compatible with the predicted environments, both natural and induced. They must also possess the rigidity and other physical characteristics necessary to insure that no deleterious coupling and no unintenional contact between adjacent components occur.

Elements of the ESP structural/mechanical subsystem are located at four remote areas of the Capsule Bus and utilize the Capsule Bus structure as the unifying element. The structural/mechanical elements as determined by the ESP equipment components are:

a. A port located at the 0° angle of attack stagnation point in the Aeroshell with provisions for mounting the total temperature sensor and the total pressure transducer and for providing gas samples for the mass spectrometer.

b. Provisions for mounting an accelerometer at the Capsule C.G.

c. Provisions for mounting the entry television cameras below the Capsule Lander footpad lower surface.

d. Provisions for mounting the UHF antenna with an unobstructed 120° field of view looking aft parallel to the roll axis.

e. A port opening into the base region with provisions for mounting the base temperature sensor and the base pressure transducer.

f. A container with provisions for mounting and insulating the ESP science instrument equipment. This container with the equipment installed comprises the ESP principal unit.

8.1.1 Stagnation Point Port - A fitting attached to the Aeroshell nose cap structure forms the moldline at the apex of the Aeroshell and incorporates two concentric ports. The tube which forms the inner port contains the total temperature sensor and provides the source for the mass spectrometer gas samples.

The outer port provides the source for the pressure sampling. The wall for the outer port provides the support for the total pressure transducer.

8.1.2 Mounting for the Accelerometer - An existing web in the Capsule Lander base platform hub fitting is utilized for mounting the accelerometer.

8.1.3 Mounting for the Entry Television Cameras - A container, insulated for heat retention, is provided for the cameras and their controls and associated electronics. The container is supported by a pyrotechnic thruster required to eject the package prior to impact. Three ball-end adjustable compression struts are provided to align and stabilize the camera container.

8.1.4 ESP Principal Unit Container - A structural container insulated for heat retention, provides two shelves to which the ESP equipment components are attached. Fittings on the container support structure provide for attachment of the UHF antenna support members.

8.1.5 Mounting for the UHF Antenna - Two struts and a bracket attached to the antenna container are bolted to the ESP equipment box to support the UHF antenna.

8.1.6 Base Region Port - A fitting attached to the UHF antenna support bracket includes a perforated tube in which the base temperature sensor is mounted. The base pressure transducer is mounted to this fitting.

8.2 DESIGN REQUIREMENTS AND CONSTRAINTS - The structural design criteria presented in Part C, Section 4 of Volume IV will be utilized to establish the loads introduced into the structural elements by the inertia effects of the ESP components. Matched hole patterns between the equipment and the support structure will be used. The thermal control provisions will meet the requirements presented in Part D, Section 3.4 of Volume IV.

8.3 PHYSICAL CHARACTERISTICS

8.3.1 Stagnation Point Port - The fitting provided at the apex of the nose cap is 9.5 inch in diameter and is made of beryllium. It acts as a heat sink to limit the temperature in the compartment behind it. It is attached to the Aeroshell structure at the same place as the altimeter antenna to utilize the inherent structural capability of the antenna for support. The fitting is attached with screws to permit removal for access to the compartment behind it. The inner port consists of a cylindrical hole and the outer port is an annular slot.

8.3.2 Accelerometer Mounting - A web at the base of the parachute catapult in the Capsule Lander base platform hub fitting provides a rigid, machined surface for mounting the accelerometer. This locates the accelerometer three inches below the C.G. on the roll axis. The Capsule environment in this locale satisfies the accelerometer's thermal requirements and no additional thermal control provisions are required.

8.3.3 Entry Television Camera Mounting - Figure 8-1 is a presentation of the camera mounting. The camera container is supported by the pyrotechnic thruster. The thruster C_L is canted outboard 30° to the vertical in the +Y reference plane. The thruster is attached by a single bolt to a fitting provided on the Capsule Lander footpad. The thruster incorporates a spherical bearing to provide omni-directional alignment capability for the cameras. Adjustable compression struts are attached to the camera container and the ball ends fit in sockets provided on the lower surface of the footpad. Insulation blankets completely encapsulate the container except for the lower surface and where the fittings extend through. A boot mounted on the Aeroshell is attached to the lower end of the container with Velcro tape. This boot prevents contaminants from depositing on the camera lenses and on the window in the Aeroshell.

8.3.4 ESP Principal Unit Container - Figure 8-2 is a presentation of the ESP Principal Unit container. Two aluminum honeycomb shelves incorporating inserts matched to the mounting holes in the equipment are provided. The lower shelf forms the base of the container. The end and top panels are permanently attached to the base while the front and back panels are made removable. Insulation completely encapsulates the container. The blankets covering the front and back are removable. Fittings, isolated from the basic structure by structural insulation blocks, are provided at the four corners of the container for mounting to the Capsule Lander base platform. Fittings are also provided for attaching the UHF antenna. The mounting provisions for the UHF antenna and for the base region port fitting are also shown in Figure 8-2. The antenna aperture protrudes through the thermal curtain. Velcro tape attaches the edge of the cutout in the curtain to the antenna container.

8.4 OPERATION DESCRIPTION - With the exception of the thruster for the entry television camera container, the elements of the ESP structural/mechanical subsystem are passive. This thruster is also passive until separation is initiated. At this time, the piston is unlocked and the container pushed away along the axis of the thruster. At the end of the stroke, the piston rod is released and stays with the container. The piston remains in the thruster, sealing it from releasing the burned gases. The compression struts drop away from the sockets as soon as the container moves and remain with the container.

ENTRY TV CAMERA STRUCTURAL/MECHANICAL INSTALLATION

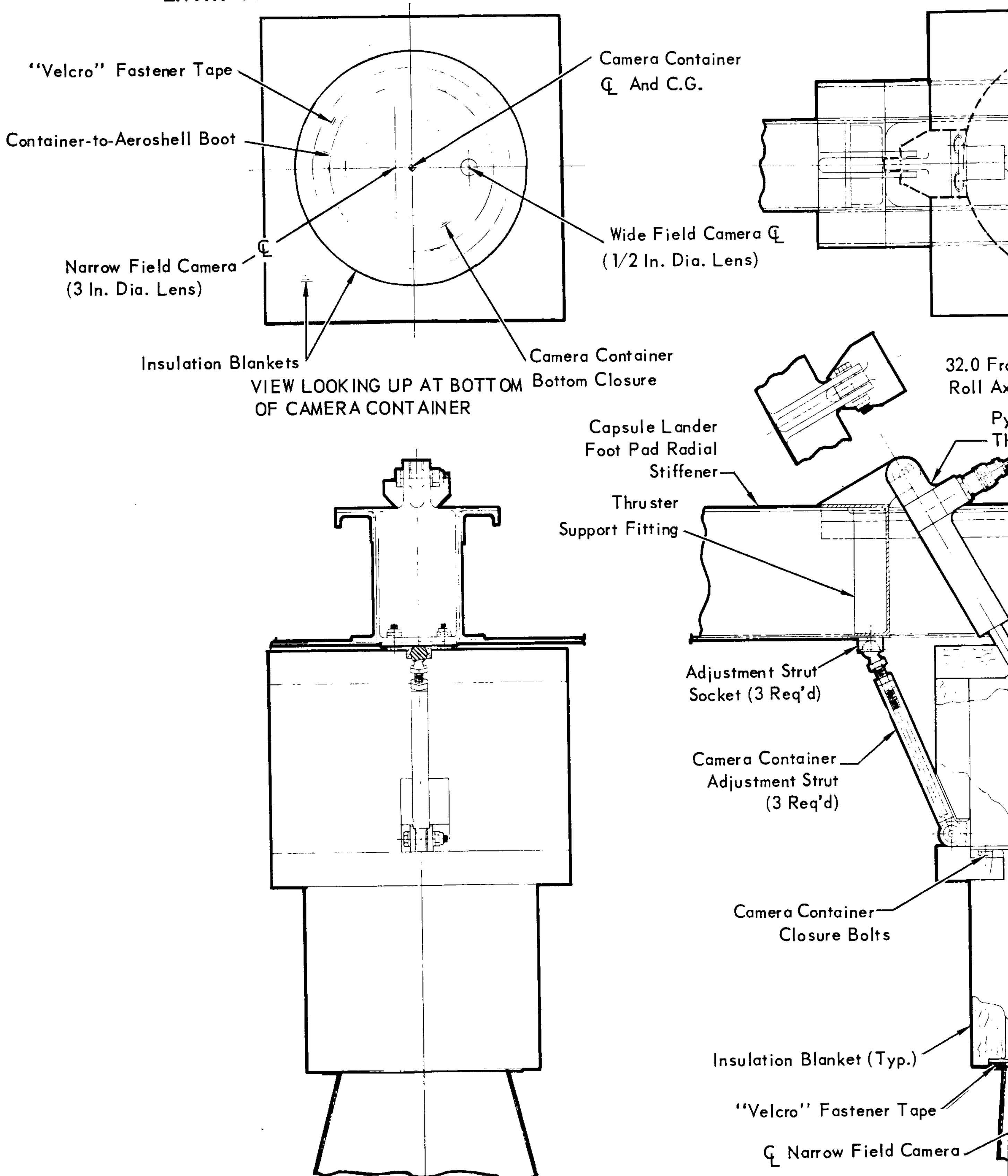
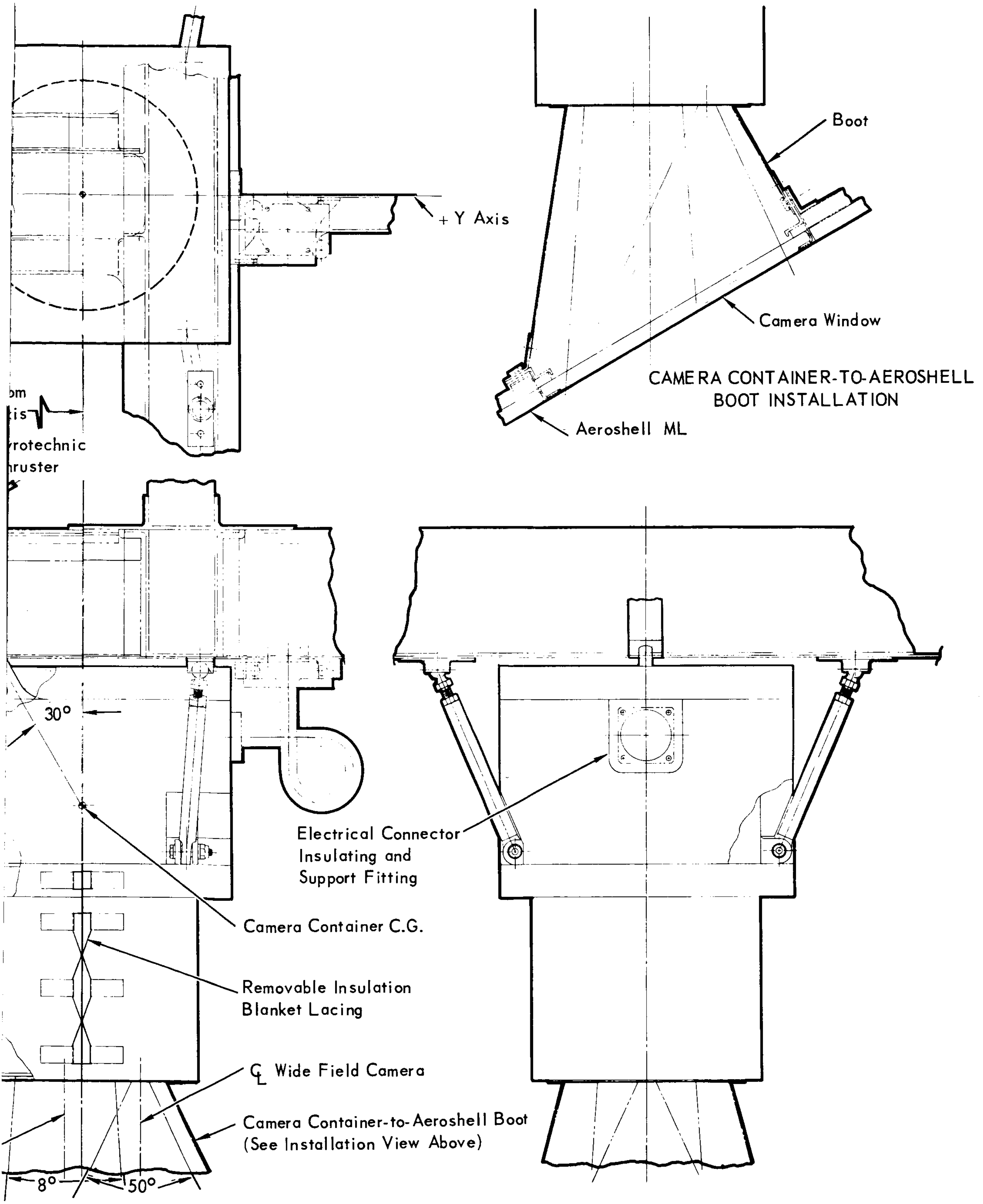


Figure 8-1


Boot
+ Y Axis
Camera Window
CAMERA CONTAINER-TO-AEROSHELL BOOT INSTALLATION
Aeroshell ML
om
tis
rotechnic
ruster
30°
Electrical Connector Insulating and Support Fitting
Camera Container C.G.
Removable Insulation Blanket Lacing
℄ Wide Field Camera
Camera Container-to-Aeroshell Boot (See Installation View Above)
8°
50°

ESP PRIMARY UNIT STRUCTURE

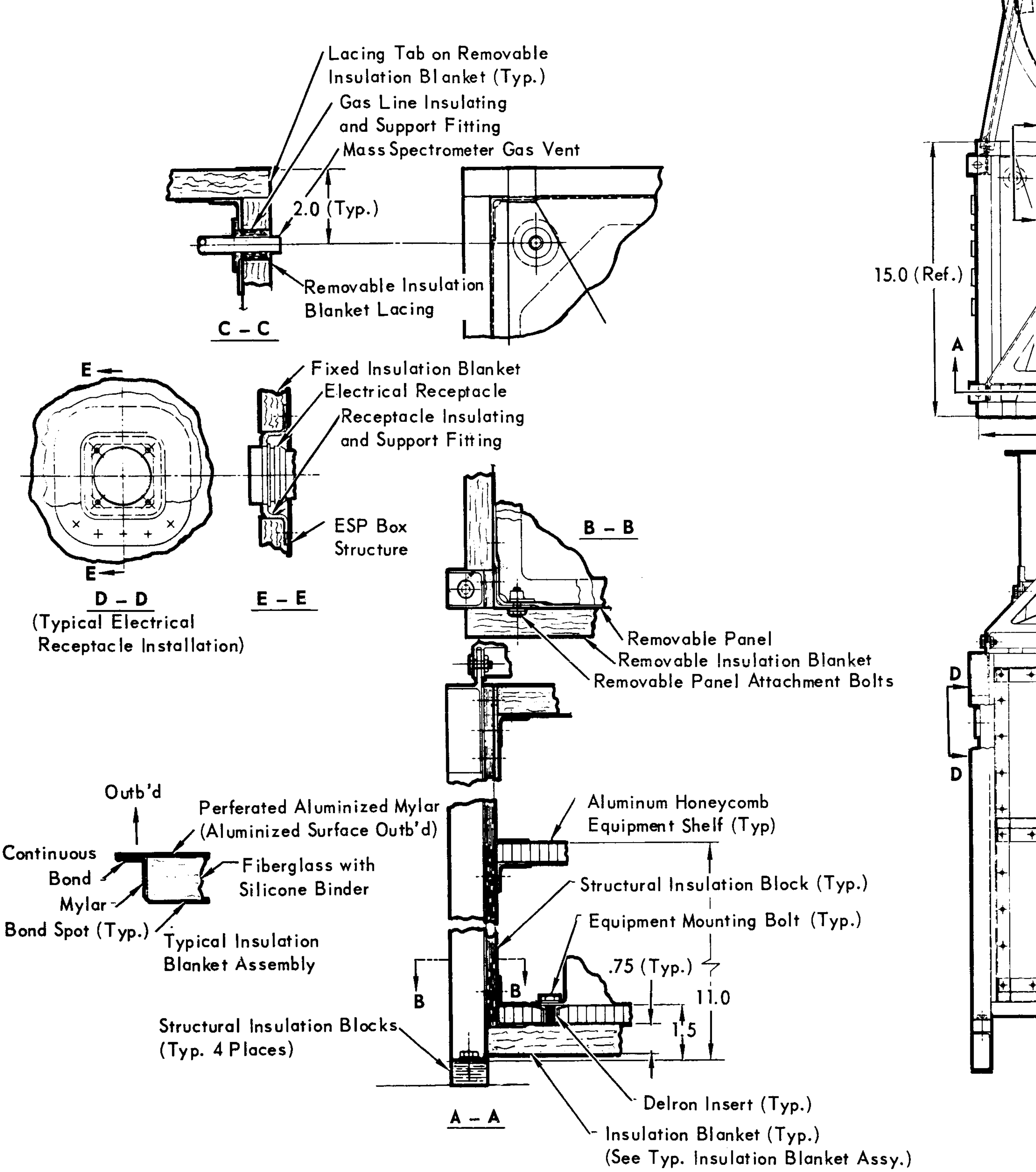
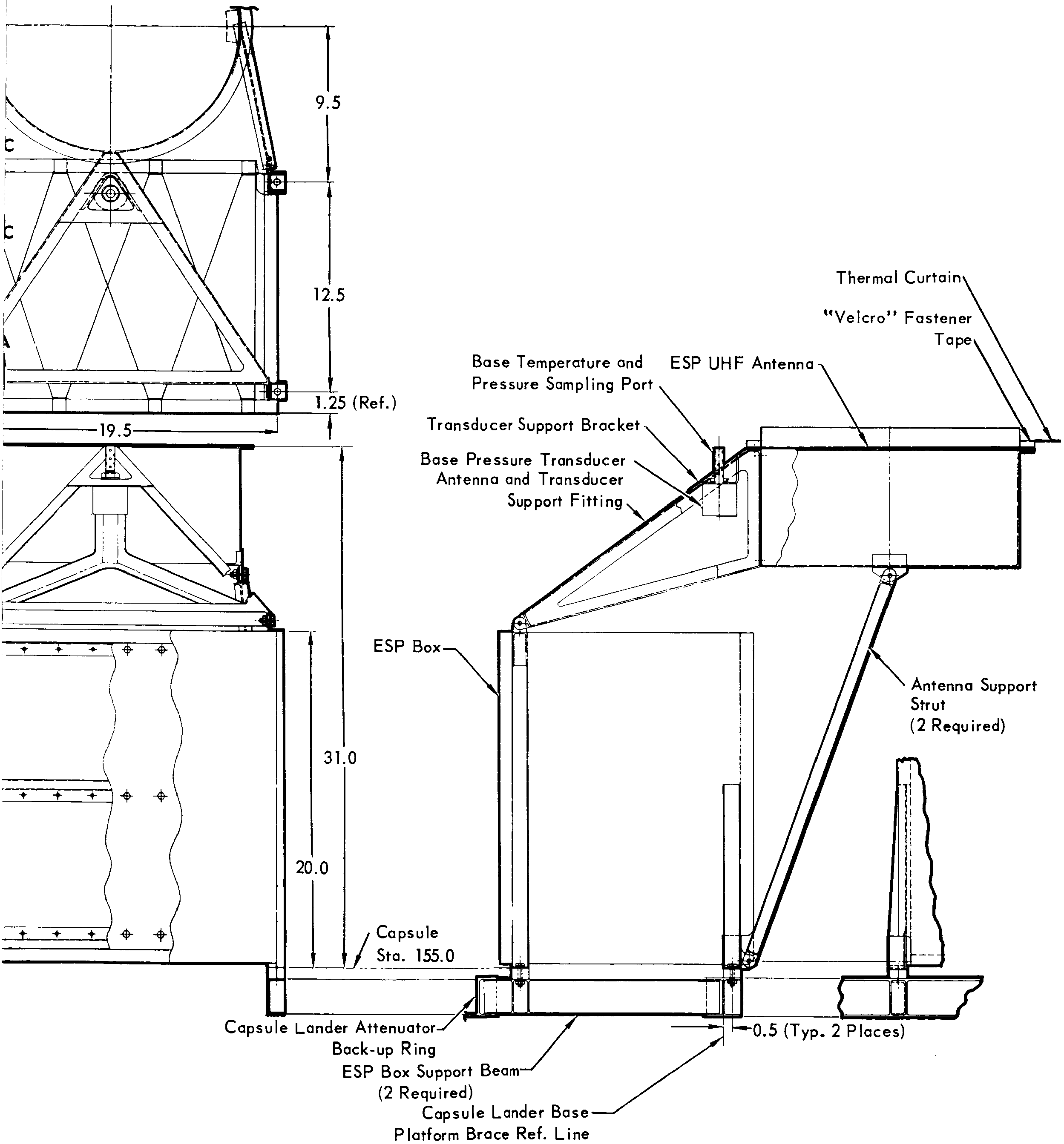


Figure 8–2


9.5
12.5
1.25 (Ref.)
19.5
31.0
20.0
Capsule
Sta. 155.0
Thermal Curtain
"Velcro" Fastener
Tape
Base Temperature and
Pressure Sampling Port
ESP UHF Antenna
Transducer Support Bracket
Base Pressure Transducer
Antenna and Transducer
Support Fitting
ESP Box
Antenna Support
Strut
(2 Required)
Capsule Lander Attenuator
Back-up Ring
ESP Box Support Beam
(2 Required)
0.5 (Typ. 2 Places)
Capsule Lander Base
Platform Brace Ref. Line

8.5 PERFORMANCE OBJECTIVES - The structural/mechanical subsystem elements support the ESP components in their required locations so that they may perform their required functions within the predicted environments. Strength, stiffness, and other physical characteristics have been considered to give assurance to this capability.

8.6 INTERFACE DEFINITION

8.6.1 Structural - The structural interface has been defined in the preceding paragraphs.

8.6.2 Thermal - The thermal interface has been defined in the preceding paragraphs.

8.6.3 Cabling - The cabling interface will consist of providing hardpoint attachments for support and cutouts for routing of wire bundles.

8.6.4 Pyrotechnic - The pyrotechnic interface has been defined in the preceding paragraphs.

8.7 RELIABILITY AND SAFETY CONSIDERATIONS - The reliability and safety considerations in the design of structure are given major attention. The philosophy used in the design and subsequent tests of actual hardware virtually negates the reliability and safety risk. This philosophy has been applied to the VOYAGER design.

8.7.1 Structure - Structural criteria are identified in terms of limit and ultimate design loads. Limit loads are those which result from the maximum expected flight or ground handling conditions. Ultimate loads are those determined by adjusting upward the limit loads by a factor of safety, normally 1.25 for flight conditions and 1.5 maximum for ground handling conditions which are potentially hazardous to personnel. The structural design is verified by tests to ultimate design loads of a representative item to provide the assurance that the reliability and safety risk is virtually eliminated and, therefore, is not approached from a statistical standpoint.

8.8 TEST REQUIREMENTS - No pre-flight, in-flight monitoring, or validation required by "no access" after sterilization tests are required for the ESP structural/mechanical subsystem.

8.9 DEVELOPMENT REQUIREMENTS - The compatibility of beryllium with the ETO decontamination cycle requires confirmation. This material is included in a current test being performed at McDonnell to determine the effect of chemical decontamination and heat sterilization on VOYAGER materials.

SECTION 9

PACKAGING AND CABLING

9.1 EQUIPMENT IDENTIFICATION AND USAGE - Packaging and cabling are the techniques of assembling components and subassemblies into assemblies to meet functional requirements and interconnecting the assemblies to provide subsystem functions compatible with mission requirements and constraints. Packaging covers the layout of internal component parts and their assembly, interconnection and installation within an enclosure and the establishing of equipment geometry to meet both internal parts and exterior assembly mounting and interconnection. Cabling covers the electrical interconnection of these assemblies of the Entry Science Package and the provision of interfaces to Operational Support Equipment and the Capsule Bus.

9.2 DESIGN REQUIREMENTS AND CONSTRAINTS - In addition to the requirements noted in VOYAGER Capsule Systems Constraints and Requirements Document SE003BB002-2A21, dated 21 June 1967, McDonnell has selected circular connectors for harness connections and established minimum wire as 24 AWG. Additional details are defined in design guide manuals and the electromagnetic control plan. McDonnell Process Specifications in effect for fabrication of electrical wire bundles are shown in Figure 9-1.

9.3 PHYSICAL CHARACTERISTICS - The primary physical aspects of the equipment packaging and cabling subsystem are indicated in Figure 9-2 for the ESP Principal Unit.

The equipment which is not location constrained to some preferred area (e.g., the entry camera, accelerometer and sensor elements) are combined in one assembly, the ESP Principal Unit. This is one compact unit in which equipment, structure, cabling and thermal control have been integrated. The equipment is installed as discrete individual "black boxes" whose form factor is established to meet functional requirements while providing for accessibility to itself, to adjacent equipment and for the internal cable harness.

The UHF antenna for the high rate radio subsystem is mounted from the main assembly by truss members and is connected by a coaxial cable to the assembly. The entry camera is mounted in near proximity with the assembly and direct connected electrically by an individual cable.

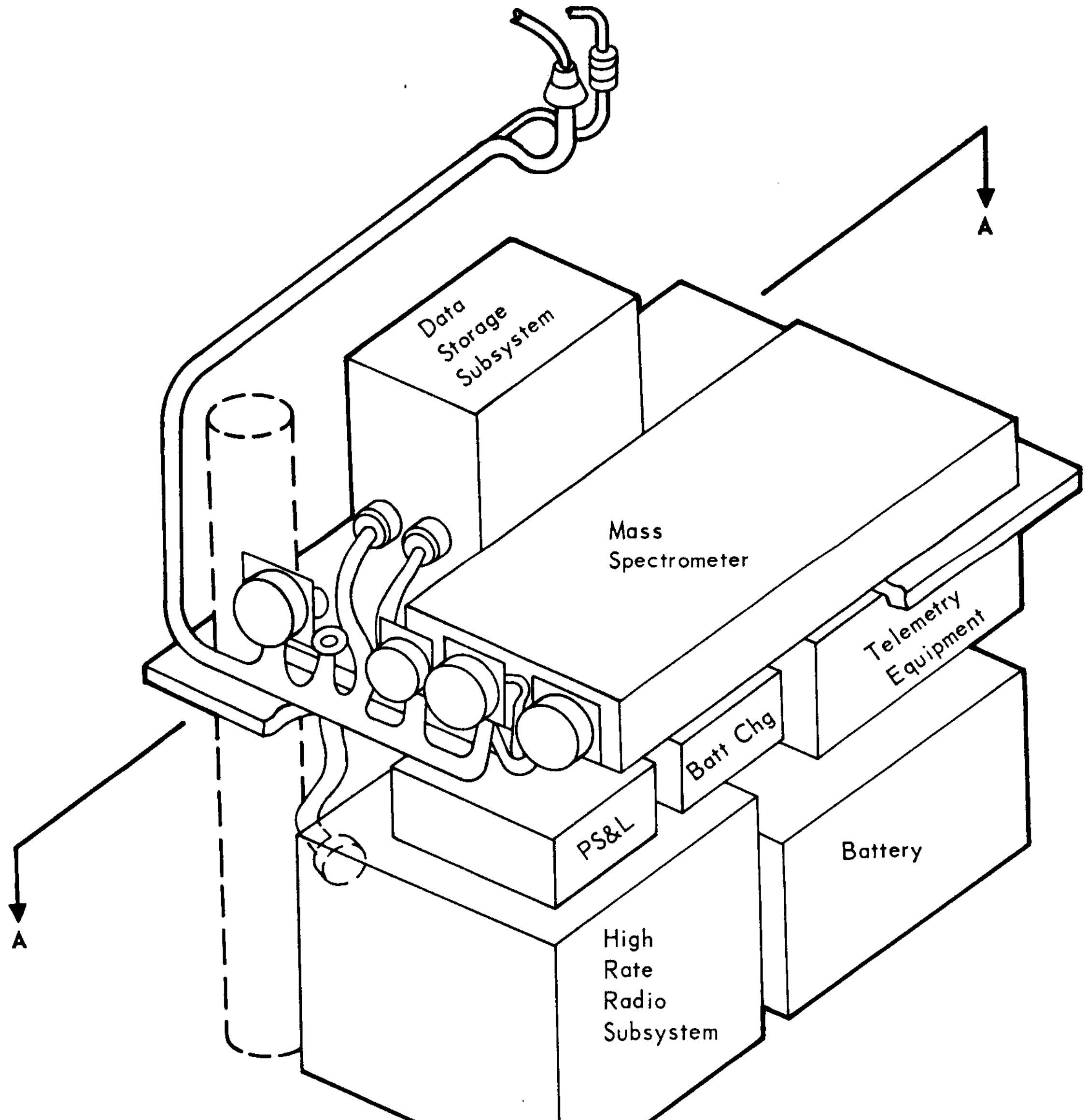
The Entry Science Package can thus be treated essentially as an entity with a few remote sensors and is capable of removal as a subsystem with minimum impact on the Capsule Bus System.

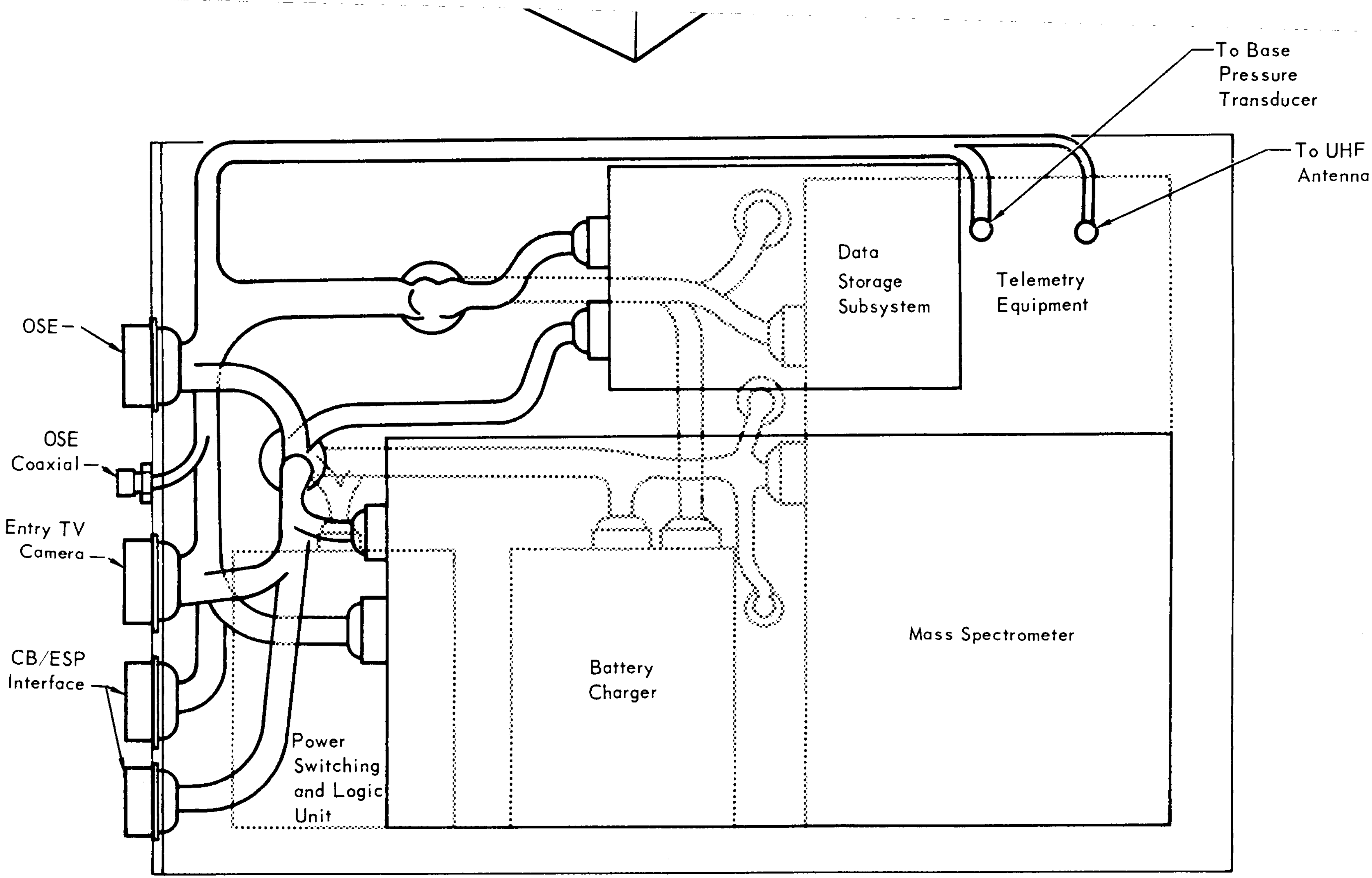
WIRE BUNDLE PROCESS SPECIFICATIONS

SPECIFICATION NO.	TITLE – DESCRIPTION
	GENERAL WIRE BUNDLE PROCESSES
PS 17400	Wiring, Electrical; Installation of
PS 17410	Wiring, Electrical, Spacecraft and Missile; Fabrication of
PS 17110	Wiring, Electrical; Identification of
PS 17111	Stripping of Electrical Wires
PS 17113	Lacing and Tying of Wiring
PS 17120	Bonding and Grounding; Electrical
PS 17153	Termination and Grounding of Shielding on Wire and Cable
PS 17172	Waterproofing of Electrical Connectors for Continuous Operation Temperatures up to 500°F
PS 17410.1	Assembly of Electrical Cable Terminals and Splices
PS 17410.2	Connectors, Electrical, Assembly of
PS 17410.3	Assembly of Radio Frequency Cable Assemblies
PS 17410.4	Connectors, Electrical, Crimp Type; Assembly of
PS 17420	Wiring, Electrical; Shielding of
PS 14070	Storage, Size Selection and Application of Heat Shrinkage Material
PS 20003	Inspection, Storage and Identification of Rubber Based Adhesives, Potting, and Sealing Materials
PS 22800	Soft Soldering of Electrical and Electronic Connectors
	COMPACT WIRE BUNDLE PROCESSES
PS 17115	Wiring, Electrical, Compact; Fabrication of
PS 17115.1	Wiring, Electrical, Compact; Twisting of Wires for
PS 17115.2	Application of Protective Jacket to Compact Electrical Wire Bundles
PS 17115.3	Wiring, Electrical, Compact; Application of Impregnating Compound
PS 17116.1	Coating of High Temperature Compact Electrical Wire Bundles
PS 17118	Wiring, Electrical, Shielded Compact; Fabrication of
PS 17118.1	Wiring, Electrical, Shielded Compact; Braiding of Shielding

Figure 9-1

ESP PRINCIPAL UNIT EQUIPMENT INSTALLATION





A–A

Figure 9–2

~~9-8~~

9-3-2

The main elements of the interconnecting cabling are MIL-C-38999 connectors and MIL-W-81381/1 wire. The connectors are potted to provide environmental protection and added wire support at the connector wire/contact transition. The individual wires are bundled into harnesses by use of lacing tapes. Wire bundle harnesses are established on the basis of signal class, physical routing, and location of the harness installation of the Entry Science Package. When a number of wires are routed together of the same classification (i.e., signals that will not cause interference to each other, but may cause electromagnetic interference to or be susceptible to other classes of signals), they may be bundled together and covered with a common shield jacket. The cables are then routed as separate cables or combined with other wires of the same signal class and bundled into a harness.

Sleeving is applied to the harnesses at local areas of possible abrasion or handling damage.

9.4 OPERATIONAL DESCRIPTION - All equipment interconnections of the Entry Science Package are made using MIL-C-38999 connectors. Manually mated miniature circular plugs are easily aligned and allow visual inspection of positive locking. During fabrication of interconnecting cables, all plugs are aligned and keyed on three-dimensional wire harness boards so that coupling to the mating equipment of the Entry Science Package requires minimum rotation of the plug prior to index keyway mating. Final coupling and locking is completed by one quarter turn of the plug coupling ring.

Requirements on the equipment for direct connection of OSE cables during subsystem test and checkout are similarly provided with MIL-C-38999 type receptacles.

9.5 RELIABILITY AND SAFETY CONSIDERATIONS - Reliability of the Entry Science Package interconnecting cable is assured by study of the requirements and constraints of each individual wire, connector and contact of each subsystem. Each step in the design of interconnecting circuits considers signal level voltage and current requirements, electromagnetic interference, mechanical strength, fabrication techniques, and routing and installation requirements of the Entry Science Package. Potential safety hazards due to high voltages associated with RF equipment are minimized by standard RF handling precautions.

9.6 DEVELOPMENT REQUIREMENTS - Components and materials required to implement high reliability equipment packaging and cabling subsystems have been selected from MAC Report E936, "VOYAGER Candidate Materials." These components and/or materials are developed into subassemblies and assemblies by progressive testing at each assembly level. Where components and/or materials not listed in MAC Report E936 are required, special testing is established to determine their compatibility with ethylene oxide (ETO) and heat sterilization.

Testing of a connector type wire and potting compounds has been initiated during Phase B to determine the effect of ETO and heat sterilization and 200 days vacuum storage. Two identical assemblies have been fabricated, using established McDonnell procedures, one to be subjected to the test environment and one for control. The latter will be stored at room temperature. In addition to the potting compatibility test results to be derived from the connector assembly testing, a potting slug containing only wires is also being tested to evaluate a third compound. The test articles are configured as shown in Figure 9-3.

One particular aspect of electronic packaging is expected to warrant considerable development effort with both basic materials and equipment design. This is the voltage breakdown problem. Since the Mars atmosphere is predicted to fall within the critical breakdown region, and the VOYAGER equipment must undergo long term space vacuum transit times, the problem must be given particular attention. There appears little chance to use previous expedients such as the hard vacuum of space or fluid (liquid and gas) filled devices to circumvent the problem. It has been concluded that the best approach is to attack the problem using insulating, encapsulating and embedding materials which have been thoroughly cured and outgassed. Good adhesion is a requisite. The standard urethane foams do not appear satisfactory, although newer products over conformally coated materials may have potential. It appears that the epoxy or silicone materials in both the unfilled and syntactic forms offer the most promise.

To a greater degree than usual, electronic packaging to eliminate voltage breakdown requires a highly integrated application of circuitry, materials and mechanical design. Each design requires review for possible breakdown areas. All design expedients such as eliminating sharp points, separating high and low voltage elements, providing adhering shields, eliminating slow bleed gas entrapping voids and providing suitable encapsulation and/or embedment will be evaluated and employed as appropriate.

WIRE ASSEMBLY TEST CONFIGURATION – ETO, HEAT STERILIZATION AND VACUUM STORAGE

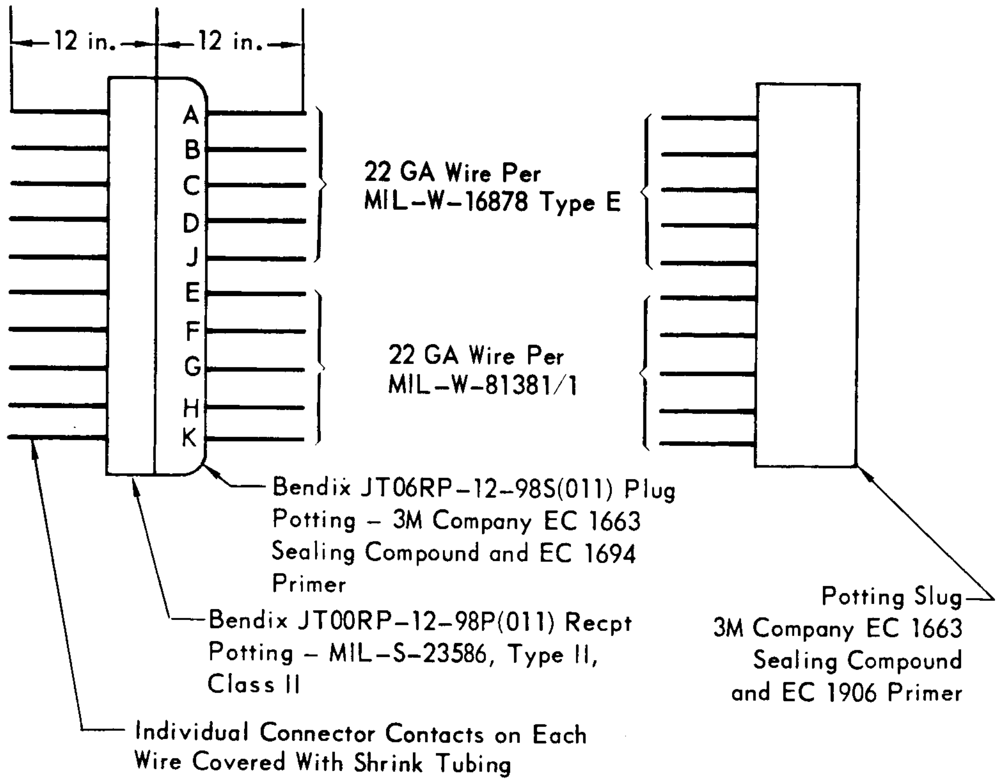


Figure 9-3

SECTION 10

THERMAL CONTROL SUBSYSTEM

10.1 EQUIPMENT IDENTIFICATION AND USAGE - ESP Thermal Control Subsystem maintains equipment temperature levels within their allowable ranges throughout all mission phases prior to landing. Temperature control is provided for both the science instruments and subsystems within the ESP Equipment Container and the Capsule Bus-mounted ESP subsystems. Until Lander separation from the Aeroshell, the subsystems operate within the overall temperature environment provided by the Capsule Bus Thermal Control Subsystem. The Capsule Bus temperature averages -140°F prior to entry and has local areas up to 800°F at the end of entry. The primary thermal control requirements for the ESP are shown in Figure 10.0-1.

The major elements of the Thermal Control Subsystem include electrical heaters, insulation and thermal control surfaces.

10.1.1 Electrical Heaters - The ESP Equipment Container and entry TV heaters supply approximately 2 and 3 watts of power, respectively, to maintain equipment temperatures during the cold, long duration interplanetary cruise. For the stagnation pressure transducer, no heater is necessary. Because of the low power level, these heaters do not require thermostats, and operate continuously when power is available from the Spacecraft. The heaters are located within the insulated region of the subsystem packages. Uniform package temperatures are obtained by providing good heat transfer characteristics from the heaters to all temperature sensitive components.

10.1.2 Insulation - The insulation, consisting of fiberglass fiber and silicone binder, has very low thermal conductivity (0.001 BTU/ft - hr - °F in vacuum), and is not sensitive to temperature excursions. This insulation is used on the subsystem packages in consideration of the following functions:

a. Minimize heater power requirement. This aids in keeping the total Flight Capsule power level below the 200 watts available from the Spacecraft during interplanetary cruise.

b. Provide contingency for later equipment design changes. The ESP temperatures could be maintained by adding more insulation, deleting the heater and using only the equipment waste heat for warming. Inclusion of a heater allows accommodation of late equipment changes by merely changing the heater size.

c. Prevent excessive equipment cooldown during powered-down periods. The Spacecraft power supply to the Flight Capsule will be interrupted during

ESP INSTRUMENT THERMAL CONTROL REQUIREMENTS

INSTRUMENT COMPONENT	NON-OPERATING TEMPERATURE LIMITS (°F)	OPERATING TEMPERATURE LIMITS (°F)
ESP Equip. Container Batteries	0 to 60	50 to 120
Entry TV Vidicons	−4 to 140	−4 to 104
Atmospheric Properties Instruments Transducer (pressure)	−328 to 256	−328 to 256

Figure 10.0-1

midcourse corrections. The insulation combined with low emittance surface coatings, prevents excessive heat loss and prevents equipment cooldown in excess of 2°F/hr during these periods.

d. Prevent excessive equipment heating from the hot Aeroshell interior surface during entry. The design structural limit temperature for the Aeroshell during entry is 800°F. The insulation and low emittance surface coating prevent this heating pulse from reaching the temperature sensitive equipment. Heat protection provisions for specific entry experiments are discussed in Part E, Section 4.5.

10.1.3 Thermal Control Coatings - The exterior surface of the insulation is coated with a low emittance gold film to assist heat retention during cruise and reflect heat input during entry. A gold coating was selected to avoid potential damage from ETO and excessive humidity during sterilization.

10.2 DESIGN REQUIREMENTS AND CONSTRAINTS - The following requirements and constraints are satisfied by the preferred design concepts:

10.2.1 Equipment Temperatures - The equipment and instruments installed in the ESP Equipment Container are maintained within a temperature range of 0° to 60°F during the inoperative period. After equipment activation, the temperatures are maintained at or less than 120°F. These temperature ranges are constrained by the ESP batteries.

Temperature limits for the entry TV and stagnation pressure transducer are shown in Figure 10.0-1.

10.2.2 Environment Temperatures - The environment adjacent to the ESP is maintained by the Capsule Bus Thermal Control Subsystem. The environmental conditions controlling ESP thermal design occur during the interplanetary cruise and Mars entry mission phases, when the nearby Aeroshell temperatures are -140°F and up to 800°F respectively. The Aeroshell temperature during entry is the structural design limit. During this period the ESP equipment is activated and must not overheat.

10.2.3 Electrical Power for Heaters - In order to maintain equipment temperatures during interplanetary cruise, electrical heaters are necessary. The total power available for all Flight Capsule equipment, including heaters, is 200 watts. This power must be apportioned between the various systems to optimize overall weight and performance balance. Thus, it is necessary to minimize heater power levels where possible without undue insulation weight penalty.

10.3 PHYSICAL CHARACTERISTICS - The ESP Thermal Control Subsystem is summarized below.

	Insulation Thickness Inches	Insulation Weight Lbs	Heater Power Watts
ESP Equip. Container	0.75	3.5	2
Entry TV	1.0	1.0	3
Transducer	1.0	0.4	

10.4 OPERATION DESCRIPTION - The ESP Thermal Control Subsystem attenuates the effects of external environment fluctuations to control the temperature of the equipment. After the initial launch transient, the Capsule Bus temperatures stabilize during interplanetary cruise to a low value (near -140°F) except in other thermally controlled areas. The ESP temperatures, without thermal control, would reach this minimum value. The combination of electrical heaters, insulation and thermal control surfaces prevents this cooldown in the ESP and maintains equipment temperatures within non-operative temperature limits.

During periods when Spacecraft power is interrupted, e.g. trajectory corrections, the insulation and low emittance surfaces help in reducing the cooldown. The insulation provides a cooldown rate as low as 2°F/hour, and prevents the instrument temperature from getting excessively low before the power supply is restored (1.5 to 3 hours without power are possible).

During the 5 minute entry period the insulation and low emittance external surface again serve to attenuate heat input from the interior of the Aeroshell.

10.5 PERFORMANCE OBJECTIVES - The ESP Thermal Control Subsystem provides a survivable temperature environment for the ESP equipments throughout all mission phases.

The following is a summary of the expected performance characteristics of the system:

ESP Equip. Container

Temperature Range	Prior to Entry: 0 to 60°F Entry: Less than 120°F
Heater Power	2 watts, continuously
Cooldown Rate (no heater power)	2°F/hour

Entry TV

Temperature Range	Prior to Entry: 0 to 60°F Entry: Less than 100°F
Heater Power	3 watts, continuously
Cooldown Rate	same as ESP

Pressure Transducer

Temperature range	Prior to Entry : -140° to 100°F Entry: Less than 200°F

10.6 INTERFACE DEFINITION - Interfaces between the ESP Thermal Control Subsystem and other subsystems are as follows:

Interfacing Subsystems/Definition

CB Thermal Control S/S	Thermal Environment established by CB
CB Power S/S	Provides heat power
CB Lander	Mechanical and thermal (mounting & rocket motors)
FC TM S/S	Monitors temperatures, heater power level

10.7 RELIABILITY CONSIDERATIONS - The reliability considerations concerning the ESP Thermal Control Subsystems were simplified by the passive nature of the system, which consists of resistance heaters, insulation, and low emittance surface coating. No dynamic components are involved. All elements of the system involve state-of-the-art materials and techniques. Thus, no serious reliability problems are envisioned for the Thermal Control Subsystem design.

10.7.1 Failure Mode, Effect and Criticality Analysis - A failure mode, effect and criticality analysis was conducted for the Thermal Control Subsystem and the results are shown in Figure 10.0-2. Inclusion of thermostats in the analysis was made to accommodate the possible design change anticipated in the future. Each failure mode is categorized according to the effect on the following mission objectives:

- o Achievement of Flight Capsule Landing
- o Performance of Entry Science Experiments
- o Performance of Landed Science Experiments
- o Retrieval of Engineering Data

The failure categories are defined as follows:

Category	Effect
1	No effect on mission objectives
2	Degrading effect on mission objectives
3	Possible catastrophic effect on mission objectives

FAILURE MODE, EFFECT AND CRITICALITY ANALYSIS
ESP THERMAL CONTROL (CRUISE)

	FAILURE MODE	FAILURE EFFECT	FAILURE CATEGORY LANDING	FAILURE CATEGORY ENTRY SCIENCE	FAILURE CATEGORY LANDED SCIENCE	FAILURE CATEGORY ENG DATA	REMARKS
Resistance Heaters & Thermostats							
Experiments (3)	Open	Degradation	1	2	1	1	Without resistance heaters, the equipment may reach degrading temperatures.
Resistance Heaters							
Proximity (1) (Continuous Duty)	Open	Degradation	1	2	1	1	Without resistance heaters, the ESP volumetric region may reach degrading temperatures

Note: Closed failure mode not applicable due to backup command shutoff capability.

Figure 10.0-2

In keeping with statements made above concerning enhancement of mission success capability, the Electric Power Subsystem FMECA (Failure Mode, Effect and Criticality Analysis) shows no category 3 failures and only possible category 2 failures. The possibility of these degrading effects is considered so remote that no further attention was allocated for the removal of these effects.

The functional logic relationship of components in the thermal control subsystem is depicted in the Reliability Diagram, Figure 10.0-3. The reliability estimate summary, Figure 10.0-4 shows the Thermal Control Subsystem calculated reliability of 0.9994. This reliability is considered conservative for the preferred design concept because of inclusion of thermostats in the analysis.

10.8 TEST REQUIREMENTS - Thermal vacuum tests to verify thermal performance of the entire ESP package are necessary before installation on the Capsule Bus. After installation, performance is verified during integrated Flight Capsule thermal vacuum tests.

ESP THERMAL CONTROL
RELIABILITY DIAGRAM

● — Resistance Heater (Continuous Duty) (1) — Resistance Heaters (3) — Control Thermostats (3) — ●

Figure 10.0-3

ESP THERMAL CONTROL
RELIABILITY ESTIMATE SUMMARY

COMPONENT	tm (1)	λ (2)	$-\ln R \times 10^6$ (3)
Proximity Resistance Heater (1) Continuous Duty	5549	0.01	55
Proximity Resistance Heaters (3)	5549	0.03	166
Control Thermostats (3)	5549	0.06	333
			554 & R = .9994

Notes: (1) tm = modified time factor x time (hr)
(2) λ = failures per million hr
(3) $-\ln R \times 10^6 = \mathrm{tm}\lambda$

Figure 10.0-4

10.9 DEVELOPMENT REQUIREMENTS - The only known development requirements for the ESP Thermal Control Subsystem is verification of sterilization compatibility for the selected insulation material. This test should be performed prior to or during Phase "C" to assure that the insulation is not affected by sterilization.